



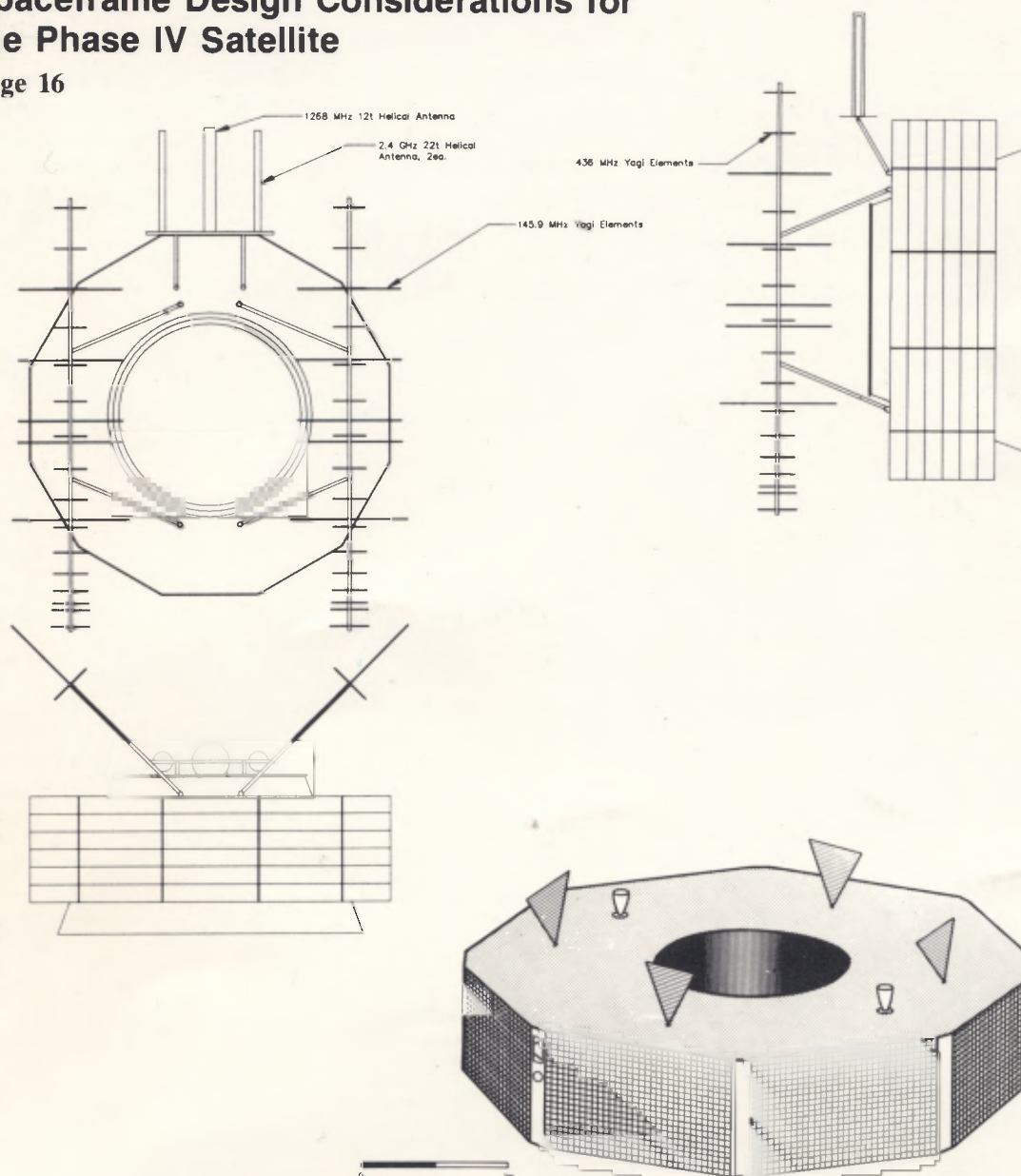
AMSAT-NA Technical Journal

Volume 1, No. 2

Winter, 1987-88

Spaceframe Design Considerations for the Phase IV Satellite

Page 16



The Radio Links to Phase III-D—An Initial System Concept

Page 23

AMSAT-NA Technical Journal

AMSAT-NA Technical Journal is a publication of the Radio Amateur Satellite Corporation of North America, AMSAT-NA. *AMSAT-NA Technical Journal* publishes papers reporting original work and significant findings in the fields of low-cost satellite design, construction, and operation, space communications, space sciences and related social value issues.

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Information for Authors and Editorial Address

Prospective authors are invited to submit articles for future issues of *AMSAT-NA Technical Journal*. Papers must contain a one paragraph abstract and be limited to ten pages (where possible) including graphs, tables and references. Preference will be given to unpublished works. Papers of general interest will be accepted for reprinting if the author includes documentation of the conditions of the original publisher for reprinting.

Material for publication should be submitted both in hardcopy form and as a text file on diskette. Any 5¼ inch format is acceptable provided the format is clearly indicated and all special character sequences used by word processors have been deleted from the text file. Figures should be drawn in a manner suitable for publication and larger than the size in which they will appear when published.

All material submitted to the Editor will be considered for publication unless otherwise requested. No material will be returned. A contribution to *AMSAT-NA Technical Journal* is welcomed from any person with an interest in the amateur radio space program. Contributions should be sent to:

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On the Cover

The cover shows possible realizations for two amateur radio satellites of the future. In the upper left is the AMSAT-NA Phase IV satellite and antenna system. Phase IV will be a geostationary satellite.

On the lower right is the Phase III-D satellite proposed

by AMSAT-DL and its position as a payload on an Ariane launcher. Phase III-D will have an elliptical orbit like Phase III-C.

Articles about the design of both of these satellites appear in this issue.

Editorial

by

Robert J. Diersing, N5AHD

Associate Professor of Computer Science
Corpus Christi State University

This second issue of *AMSAT-NA Technical Journal* realizes one of the goals President Riportella and I agreed should be accomplished by its publication—that of informing the satellite user of the variety of engineering and construction activities required to produce an operational amateur radio satellite. In this issue we bring you information about spacecraft yet to be built: Phase IV, Phase 3D, and PTSE-H; and those that have already been built: Phase 3C, Phase 3B, and NUSAT I.

Amateur radio satellite enthusiasts are anxiously awaiting the launch of AMSAT Phase 3C in early summer. Two articles describe subsystems used in Phase 3B and, with various improvements, in Phase 3C: “The Integrated Housekeeping Unit — A Method of Telemetry, Command and Control for Small Spacecraft” by Gordon Hardman and “The Propulsion Systems of the Phase III Series Satellites” by Dick Daniels. Both of these papers were presented at the First Utah State University Conference on Small Satellites held on October 7, 1987.

From a user interest viewpoint, the AMSAT-DL RUDAK experiment on Phase 3C should prove to be very popular. Two articles related to RUDAK appear in this issue: “Some Thoughts on RUDAK Traffic Control” by Hanspeter Kuhlen and “PSK Interface for the TAPR TNC-1” by Peter Gulzow. These articles have appeared in the *AMSAT-DL Journal*.

The satellite design category is adequately represented in this issue as well. “Spaceframe Design Considerations for the Phase IV Satellite” by Dick Jansson describes various factors of spaceframe design, especially those affecting thermal design for the Phase IV geostationary satellite proposed by AMSAT-NA. “The Radio Links of Phase III-D: An Initial System Concept” by Karl Meinzer shows how AMSAT-DL intends to improve the Phase III series to serve a larger user community. Finally, we have “A Method for Evaluating Antennas for a Low Earth Orbit Mission” by Courtney Duncan showing factors being considered during the design of PTSE-H (also called HOUSAT).

Two more articles round out this issue and each is important for different reasons. First, “Spacecraft Technology Trends in the Amateur Satellite Service” by Dick Jansson serves to remind us of the history and accomplishments associated with our chosen specialty within Amateur Radio. This paper was also presented at the Utah State University conference mentioned earlier and as a result served to present those accomplishments to a wider segment of space technology professionals. Second, “NUSAT I’s ‘Layered’ Protection Software Design” by Chris Williams is important because we have not seen much information in print about small satellite projects outside the amateur radio community. Chris’s article also presents a different philosophy of user access to the spacecraft onboard computer.

In any all-volunteer effort such as the production of this journal, it is always important to acknowledge those who contributed to the project. I would like to express my appreciation to those authors who prepared articles especially for this issue: Dick Jansson, Chris Williams, and Courtney Duncan. Dick Jansson also provided the drawings for Phase IV shown on the cover. Thanks are again due AMSAT-DL for providing English translations for the articles reprinted from the *AMSAT-DL Journal*.

On the production side of the house are my secretary Maggie Hernandez and my wife Julie doing some of the word processing and proofreading. Typesetting and graphic design were once again done by John Stalmach, CCSU Office of Public Information. I was grateful to receive the Publications Achievement Award at the last AMSAT-NA annual meeting. However, it should have been given to those people mentioned here and in the previous editorial.

The next issue of *AMSAT-NA Technical Journal* will be published this summer. With all of the design and construction activities now in progress I hope I will be overwhelmed with articles.

Spacecraft Technology Trends in the Amateur Satellite Service

by Dick Jansson, WD4FAB
Phase IV Study Team

(Presented at the 1st. Utah State University
Conference on Small Satellites, October 7, 1987)

ABSTRACT

Small communications satellites have been employed by the Amateur Radio community for over twenty five years. These satellites have been used by tens of thousands of radio amateurs for recreation, education and scientific investigation. The Amateur satellite program today is international in scope with nine countries having contributed space flight hardware to the overall effort. A total of 14 satellites have been fabricated in the OSCAR series. Another 11 have been constructed in other programs resulting in a total of 25 satellites and 29 spacecraft-years of orbital experience.

This paper will address the history of the spacecraft developed for the Amateur Satellite Service with emphasis on technology trends in the program. Particular emphasis will be placed upon the mass, cost, system capability and construction phase duration for each of the fourteen satellites of the OSCAR series.

The Amateur satellite program has not simply adopted technology from other larger satellite programs. In many cases entirely unique design concepts were employed, required because of the small size of the spacecraft, the very limited power available or the limitation of fiscal resources. The development of high efficiency linear communications transponders and the design of highly integrated command, control and telemetry equipment are two examples. For many years these techniques have been of little value to designers of larger spacecraft, who have alternatives. With the new interest in lightweight satellites at low cost, these techniques may take on renewed applicability.

Introduction

Small communications satellites have been employed by the Amateur Radio community for over twenty-five years. These satellites have been used by tens of thousands of radio amateurs for recreation, education and scientific investigation. In recognition of the potential value of these activities, the International Telecommunications Union established, in 1971, the Amateur Satellite Service; a separate service from the Amateur Radio Service but with common objectives. The Federal Communications Commission has followed suit in its Rules and Regulations, Part 97, Subpart H. Spectrum has been allocated to the Amateur Satellite Service throughout the HF, VHF, UHF and microwave bands.

The Amateur satellite program today is international in scope with nine countries having contributed space

flight hardware to the overall effort. A total of 25 satellites have been fabricated in the world-wide Amateur Satellite program resulting in 29 spacecraft-years of orbital experience to date. One of the satellites failed to achieve orbit due to a launch vehicle failure and one awaits launch on the new European ARIANE-4.

This paper will address the history of the spacecraft developed for the Amateur Satellite Service with emphasis on technology trends in the program. Particular emphasis will be placed upon the mass, cost, system capability and construction phase duration for each of the fourteen satellites of the OSCAR series. The Amateur satellite program has not simply adopted technology from other larger satellite programs. In many cases entirely unique design concepts were employed, required because of the small size of the spacecraft, the very limited power available or the limitation of fiscal resources.

The development of high efficiency linear communications transponders and the design of highly integrated command, control and telemetry equipment are two examples of these technologies. For many years these techniques have been of little value to designers of larger spacecraft, who have alternative technology options. With the new interest in lightweight satellites at low cost, these techniques may take on renewed applicability.

In the process of developing spacecraft of this nature, new strategies for environmental testing, component selection, materials selection, system redundancy and program management had to be developed. Some of these new strategies are driven by fiscal restraints, while others by manpower resources, these approaches differ significantly from conventional spacecraft programs.

Historical Background

The launch of the first artificial satellite, Sputnik I, excited the imaginations of a great many people around this globe. The world of Amateur Radio ("ham" radio) is one that historically has attracted devotees that have been filled with curiosity, investigation and "imagineering," all to create new methods and applications in the realm of radio communications. These two statements, seemingly unconnected, converged about two years following the launch of Sputnik I, with a group of California amateurs, incorporated as Project OSCAR, initiating a program to build an OSCAR (Orbital Satellite Carrying Amateur Radio). After two years of their efforts, OSCAR I was launched on 12 December

1961 as a "piggy-back" secondary payload aboard an Air Force Thor-Agena launch vehicle carrying the Discovery XXXVI payload. These events have been suitably documented by Davidoff and others, starting with the visions of a magazine writer.[1]

Oscar I was but the first of some 25 satellites, built by radio amateurs around the world, that have either been orbited or are planned to be launched. Fig. 1 chronicles the relentless pursuit of radio amateurs toward achieving highly reliable global communications with complete freedom from the effects of ionospheric propagation phenomena. The 40 years of past and future OSCAR activity shown provide the background of our experience and lay the plan for our future expectations. Fig. 2 is a log of the operating history of each of these satellites, presented with a logarithmic abscissa covering time from 1.2 days to 31.7 years. Note also that there are eleven Russian radio amateur satellites that have been launched, the RS and Iskra series. As we have very little information on their construction and properties, further efforts to include them in this discussion is not practical.

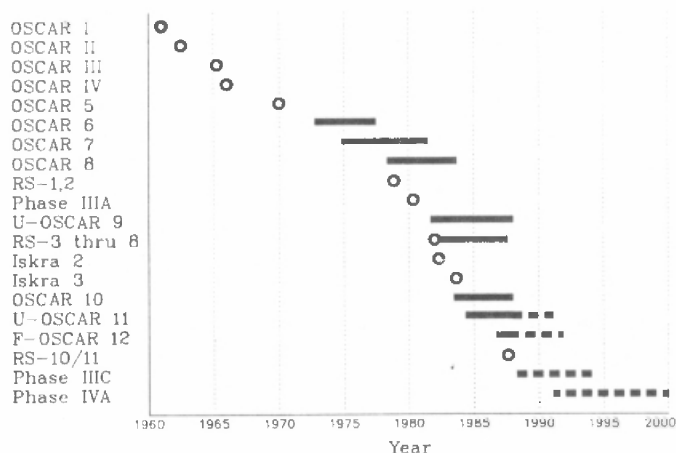


Figure 1. Amateur Satellite Flight History

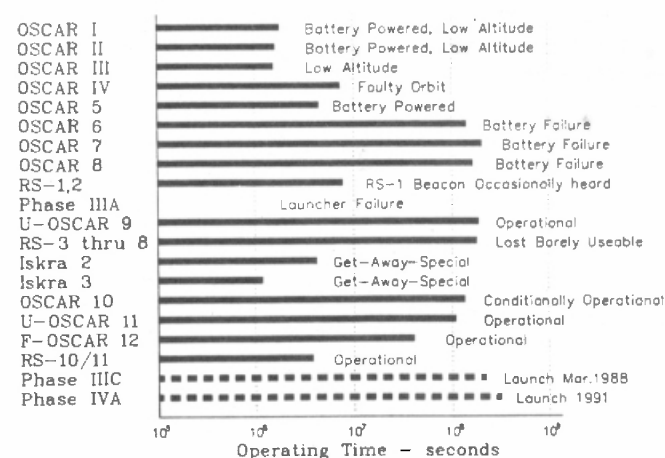


Figure 2. Amateur Satellite Flight Log

AMSAT

Starting with OSCAR 5, a new organization was formed for the exploitation of amateur radio satellites. Called

the Radio Amateur Satellite Corporation, it is more commonly known as AMSAT, a successor to Project OSCAR in launching amateur satellites.

AMSAT has grown into an international organization and spun off a number of affiliate organizations in other countries. Most of the work done on amateur satellites in the last fifteen years has been done as international efforts, with one or more national groups defining the basic spacecraft. This consortium has also provided the systems design and control and defined the subsystem interfaces. Substantial design flexibility exists in the subsystems, as long as their interface requirements are met, and the execution of these subsystems have been delegated to even other groups. Phase III satellites provide a good example of this process.

The central consortium has been between (as we know it now) AMSAT-NA (AMSAT North America) and our West German colleagues, AMSAT-DL. Subsystems have been fabricated by Bulgarians, Japanese, Australians, South Africans and other national AMSAT groups. Even the spaceframe assembly in suburban Washington, DC, looked like a small United Nations.

This decentralized, all volunteer army does have its drawbacks in managing a program, but the dividends are that the program can draw on the talents of highly capable and motivated persons. Many of these volunteers are aerospace professionals on their own right, but the aura of an amateur spacecraft attracts them to contribute their time and talents to the program.

OSCAR Program Phases

The many spacecraft constructed by radio amateurs can be roughly classified by their intended function into four Phases. Phase I designs comprise low earth orbit (LEO), short lifetime beacon satellites, such as OSCARs I, II, III, 5 and Iskra 1 and 2. Phase II designs are also LEO (but not as low an altitude), long lifetime satellites with active transponders and experiments, such as OSCARs 6, 7, and 8, UoSAT OSCARs 9 & 11 and RS 1-11. Phase III satellites are designed to function in elliptical Molnia-type orbits at high altitudes, for long lifetimes and with wide area transponder coverage. Examples of Phase III satellites are the ill-fated Phase IIIA, OSCAR 10, and the soon to be launched Phase IIIC. Phase IV satellites are now in the study and design phase and crown this development cycle with geosynchronous "constant" position orbits providing 24 hour/day communications over nearly half the globe per satellite.

OSCAR IV was ahead of its time by 2.5 decades with a Phase IV mission, that we now plan for the early 1990s.

OSCAR Satellite Characteristics

As would be expected in a progression of satellite designs covering 2.5 decades, substantial advancements have been made in the features and capabilities of the OSCARs. Correspondingly, satellite mass, cost and complexity have increased. OSCARs I and II were literally

assembled in California home workshops in the originator's garages. Even substantial elements of OSCAR 10 were concocted in home workshops and kitchen ovens, although that spacecraft was too large for a normal home workshop and required a more formal assembly laboratory. The Phase IVA spacecraft will be just too immense to place in anyone's shop, much less get through the doorways of all but the most special of commercial buildings. These size examples only illustrate our progress.

Fig. 3 illustrates this spacecraft growth from OSCAR I to Phase IVA, with the increase of nearly 100 (actually 19.5 dB, for those who think in logarithmic terms) in spacecraft orbital mass. Similarly, power consumption, in our own miserly power budget terms, has grown by two to three decades, as shown in Fig. 4, although that data is considerably skewed by mission objectives and the available power sources employed.

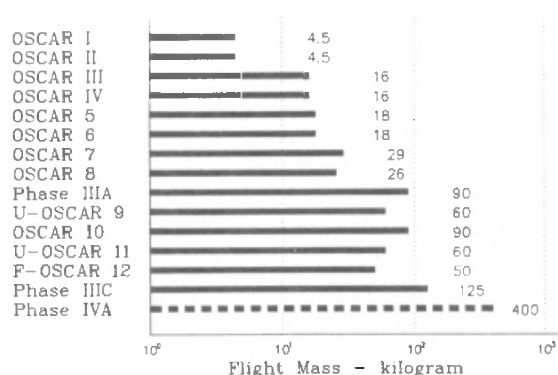


Figure 3. Amateur Satellite Flight Mass

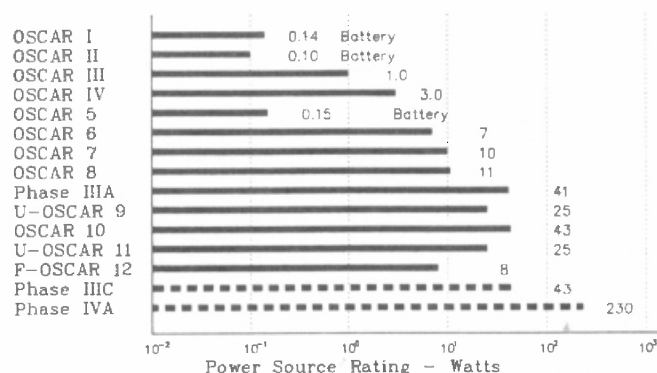


Figure 4. Amateur Satellite Power Rating

OSCAR Payloads

To further grasp the degree of evolution of the OSCAR satellites Fig. 5 shows several measures of the capabilities of these missions. Measures are in terms of the relative sophistication employed for information telemetry, control capabilities and transponder highest frequencies. In recent years the degree of control cannot be related to the number of channels employed as microprocessor on-board-computers (OBC) have changed that meaning, instead the measure is given in terms of the RAM memory carried aboard.

While more than 25 years ago we spoke about just being able to copy the single channel CW beacon of OSCAR I, these days, even in LEO satellites (Fuji OSCAR 12) we speak in terms of copying the packet bulletin board system (PBBS) messages and leaving stored messages for other amateurs at other locations on this globe. This FO-12 activity has been witnessed, and viewed in awe, as this paper was being prepared. Meanwhile, the AFC locked phase-shift-keyed (PSK) modem automatically tracked the down-link information through all of its signal Doppler shift, and the computer kept right on flashing all of the traffic onto the screen as the satellite passed.

In Fig. 5 the information regarding highest frequency is a bit misleading, as while OSCAR 7 had a 2304 MHz beacon aboard, it could not be turned on for legal reasons, dictated by the FCC. For Phase IIIC and Phase IVA we are planning to employ this kind of 13 cm equipment in real transponder service. The principal transponder of Phase IV will be several band segments (separate transponders) of 24 cm uplink with 13 cm downlink, fairly advanced even for many of today's amateurs, but a very real need for that mission. These communications links are feasible today, and will be common place at the time of the expected Phase IVA launch.

This advancement in communications transponder capabilities is more suitably illustrated in Fig. 6, showing the total compounded bandwidths of all transponders aboard the respective satellites. While the total bandwidth for OSCAR 10 looks large, and it was actually flown with that capability, a single transistor failure in the 800 kHz wide Mode L (24 cm uplink, 70 cm downlink) transponder reduced the gain and operationally effective bandwidth of that unit to about 100 kHz, making the real total for OSCAR 10 about 270 kHz. The upcoming launch of Phase IIIC, with 500 kHz bandwidth presents a real improvement over predecessors, and the 1.22 MHz bandwidth for Phase IV represents a substantial improvement over Phase IIIC. Most of the transponders have been linear translators, faithfully retransmitting their input signals. Some recent transponders have deliberately not been linear. The digital transponding functions of the DCE of the UoSAT OSCAR 11 and the Phase IIIC RUDAK packet system are examples.

	Telemetry Channels	Command Channels	Computer Memory kByte	Highest Frequency MHz.
OSCAR I	1	0		144
OSCAR II	1	0		144
OSCAR III	2	0		145
OSCAR IV	2	2		432
OSCAR 5	7	2		144
OSCAR 6	24	21		435
OSCAR 7	96	70		2304
OSCAR 8	6	5		435
Phase IIIA	64	OBC	16	435
U-OSCAR 9	105	OBC	32	10 GHz
OSCAR 10	64	OBC	16	1270
U-OSCAR 11	156	OBC	368	2400
F-OSCAR 12	62	40	48	436
Phase IIIC	64	OBC	32	2401
Phase IVA	~128	OBC	~640	2402

Note: OBC is On Board Computer

Figure 5. Amateur Satellite Capabilities

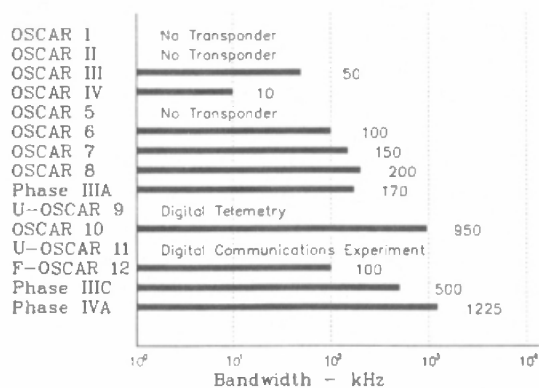


Figure 6. Compounded Transponder Bandwidth

To conclude this discussion on the prime payloads of the OSCAR satellites, the communications transponders, a useful measure of capability is shown in terms of a form of "gain-bandwidth" product. Fig. 7 shows our version as a "power-bandwidth" product, EIRP-bandwidth in Watt-kiloHertz, to be specific. This measure accounts for transmitter antenna gain, transmitter power and bandwidth, all expressed as a single product. It can be seen that the plans for Phase IVA present a stupendous decade growth over even the inflated value for OSCAR 10. Not normalized in this process, however, are the mission requirements. The lower EIRP-B.W. for Fuji OSCAR 12 does not demean its performance, as it is a long-life LEO satellite and produces quite strong signals despite its lower transponder output. Conversely, the Phase III and IV satellites need higher powers and antenna gains to provide usable signals from their 36,000 km altitudes.

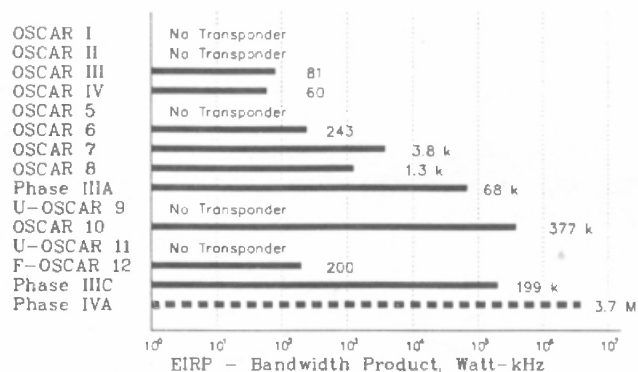


Figure 7. Amateur Satellite Transponder System Performance

Attitude Control and Station Keeping

As the missions of the OSCAR satellites have become more sophisticated, so too have the methods employed for attitude control. Transponder antennas, even fairly simple ones, have directivity characteristics, and spacecraft attitude control is important to maintain useful communications links. Fig. 8 shows this progression of spacecraft control, running from no control on

the earliest, to simple bar magnets, and further on to a complex computer controlled magnet system for active spin and attitude control. This last system is used on the Phase III satellites and senses Earth and Sun positions, computer processing the data and controlling three sets of magnets for spacecraft spin and attitude. [2] The UoSAT program also uses magnetic torquing, but in conjunction with gravity gradient booms and magnetometry for attitude sensing. The Phase IV spacecraft will employ transponders with highly directive antennas that will require very precise attitude control in the body stabilized mode at geosynchronous altitudes. We hope to employ a simple, low cost reaction control system for this Phase IV mission.

	Orbit Type	Attitude Control
OSCAR I	LEO	None
OSCAR II	LEO	None
OSCAR III	LEO	None
OSCAR IV	GTO	Spin
OSCAR 5	LEO	Spin, Passive Magnets & Lossy Dampers
OSCAR 6	LEO	Spin, Passive Magnets & Lossy Dampers
OSCAR 7	LEO	Spin, Passive Magnets & Lossy Dampers
OSCAR 8	LEO	Spin, Passive Magnets & Lossy Dampers
Phase IIIA	Molnia	Spin & Computer Operated Magnets
U-OSCAR 9	LEO	Gravity Gradient Boom
OSCAR 10	Molnia	Spin & Computer Operated Magnets
U-OSCAR 11	LEO	Gravity Gradient Boom & Active Magnets
F-OSCAR 12	LEO	Spin, Passive Magnets & Lossy Dampers
Phase IIIC	Molnia	Spin & Computer Operated Magnets
Phase IVA	Geop	Earth Oriented Body Stabilized With Reaction Control System

Figure 8. Amateur Satellite Attitude Control

As the attitude control needs have become more demanding, the microprocessor computer has made meeting these demands feasible. Pointing requirements have dictated that we are able to measure our position in space and the location of the Sun and Earth. While these needs are not at all new to the space industry, some of the solutions employed and proposed are unique. It should be noted that the space industry probably has as many solutions for position determination as there are satellites; this kind of measurement has had no really universal solution. Suffice, there are probably very few missions, even of the low-cost small satellite field that will not require some sort of attitude control, even passive.

OSCAR Satellite Costs and Effort

The discussion to this point has generally been on very quantifiable terms of satellite design and performance. Entering the arena of the cost of an OSCAR satellite and how long it took to build become subjects that are difficult to grasp. One must first understand that while AMSAT's satellite builders may be well paid and respected professionals in their own diverse fields, they are truly volunteers when it comes to building OSCAR satellites. Evaluating a fair-market-value for the labors of tirelessly applied volunteer efforts is next to impossible. Further, many companies in the aerospace industry knowingly contributed to the programs in many diverse ways, such as authorization of computer time

resources for satellite design efforts, specialized components and many other countless contributions.

Fig. 9 shows that while we have assembled satellites for very nominal monetary amounts of out-of-pocket funds, the trend of costs are escalating nearly five decades while the mass only grew two! These increases all despite the application of innovative solutions to normally expensive problems. One cost that is now becoming substantial is that of a launch position. Most of the early OSCARs were launched for no fees at all, as there were usually excess launcher capacity available, or the launcher was new and experimental, as was the case of the ARIANE that was to launch the Phase IIIA spacecraft in 1980. The loss of that launcher was a hardship to the ESA program, and an absolute disaster to the amateur satellite program. As launching facilities have become more experienced and launches more routine we have been expected to underwrite some share of the integration and launch costs. This trend is expected to continue as space programs mature.

Another impact to the cost picture is illustrated by the OSCAR 10 effort. A substantial amount of the work needed was done in duplicate (e.g. the spaceframe) and this subsequently reduced the costs for its later sister spacecraft, Phase IIIC.

Fig. 10 gives some insight to the elapsed calendar time for the several OSCAR programs. Buried in this information is the trend that second and third models of a particular program have required less time to achieve, not a terribly surprising situation. The experience factor has been substantial; hence the Phase IIIA effort was 5.5 years, OSCAR 10 was 2.5 years, and Phase IIIC only two years. A two year program seems to be the minimum, although some programs, through the application of super-human efforts have produced complex spacecraft, such as UoSAT OSCAR 11, in as short a period as 6 months.

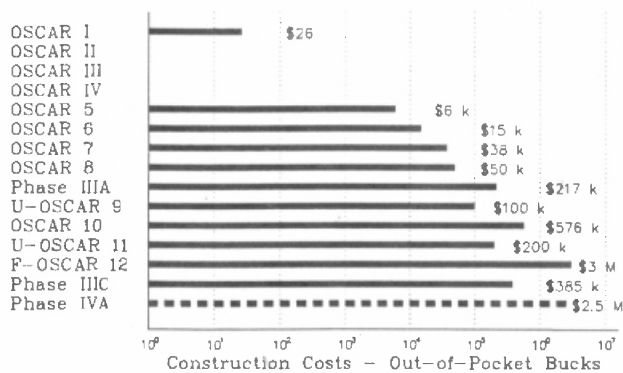


Figure 9. Amateur Satellite Costs

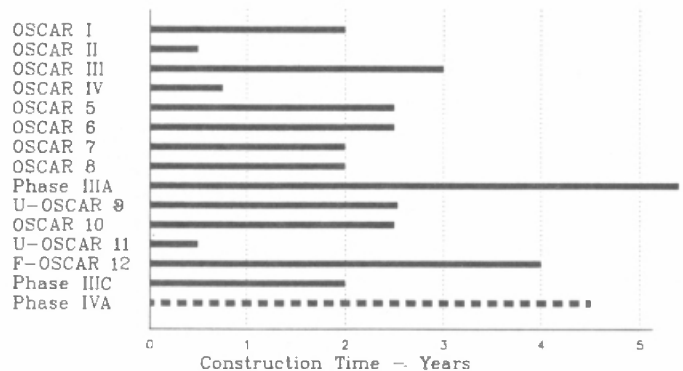


Figure 10. Amateur Satellite Construction Time

OSCAR Program Execution

This subject of constructing complex hardware projects with mostly volunteer labor cannot be left without commenting on the immensity of the differences in the management techniques that need to be applied. AMSAT's procedure may seem harsh and leaderless to some: the process really sorts out those individuals who will perform with a major amount of self motivation, and those who are just along for the ride. This latter group does not last very long on an AMSAT program. This yoke of volunteerism places demands on an AMSAT program manager that pale those placed on a manager of an all professional effort.

Technical management on a low cost spacecraft program also requires inventiveness in the process of evaluating the suitability of components to do their job. Decisions also must be made on just how much testing of various system blocks is needed to provide a flight worthy confidence, without the cost overkill of a NASA or DOD program.

Component testing varies from a full stress and test burn-in, to a simple value check measurement at assembly. The drivers on deciding these criteria are the confidence in the class of component and its failure mechanisms. In some cases, by purchasing MIL-STD components, there is no individual component testing done at all, except at the subassembly level.

Testing of subassemblies primarily are just those of an extensive room temperature burn-in and functional test to sort out component and circuit infant mortalities. Environmental testing is relegated to the overall spacecraft integration level of assembly. The success record provides some substantiation to these philosophies. Useful life termination of early spacecraft, such as OSCAR 6, 7 and 8, have been related to high battery temperature problems. This is a thermal design problem that has been solved on later programs by obtaining the services of qualified personnel in the design phases of the program, long before the fact of the failure mode.

Conclusions

AMSAT's proven track record of successful spacecraft has given the confidence that we can build a geostationary class of satellite. We have to be very careful, however, as the size, cost and complexity of this kind of program may grow beyond that which can be handled as a volunteer program constructing a "small satellite."

Acknowledgment

I would like to thank Jan King, Vice President for Engineering, AMSAT-NA, for the opportunity to practice some of the really fun aspects of my profession over

the last decade; opportunities that were not available to me in my vocational pursuits. Further, this association with Jan has followed me into retirement, providing me the challenge and fun of staying abreast of spacecraft technology, all of which is gratefully appreciated.

References

1. Martin Davidoff, *The Satellite Experimenter's Handbook*, The American Radio Relay League, Newington, CT, 1984.
2. Gordon Hardman, "The Integrated Housekeeping Unit," 1st USU Conference on Small Satellites, Logan, UT, 1987.

Keplerian Elements for Amateur Radio Satellites

Satellite: oscar-9

Catalog number: 12888

Epoch time: 88027.03662562
Element set: 145
Inclination: 97.6313 deg
RA of node: 55.7850 deg
Eccentricity: 0.0002987
Arg of perigee: 119.6760 deg
Mean anomaly: 240.4771 deg
Mean motion: 15.31271780 rev/day
Decay rate: 4.039e-05 rev/day²
Epoch rev: 35097

Satellite: oscar-10

Catalog number: 14129

Epoch time: 88024.15159285
Element set: 326
Inclination: 27.4239 deg
RA of node: 341.4037 deg
Eccentricity: 0.6027238
Arg of perigee: 275.0069 deg
Mean anomaly: 23.0978 deg
Mean motion: 2.05877787 rev/day
Decay rate: 8.7e-07 rev/day²
Epoch rev: 3471

Satellite: oscar-11

Catalog number: 14781

Epoch time: 88021.24452104
Element set: 286
Inclination: 98.0786 deg
RA of node: 87.5337 deg
Eccentricity: 0.0013575
Arg of perigee: 129.7470 deg
Mean anomaly: 230.4924 deg
Mean motion: 14.62220848 rev/day
Decay rate: 2.76e-06 rev/day²
Epoch rev: 20756

Satellite: oscar-12

Catalog number: 16909

Epoch time: 88018.24549393
Element set: 79
Inclination: 50.0155 deg
RA of node: 84.2716 deg
Eccentricity: 0.0011218
Arg of perigee: 113.5298 deg
Mean anomaly: 246.6712 deg
Mean motion: 12.44394600 rev/day
Decay rate: -2.5e-07 rev/day²
Epoch rev: 6516

Satellite: rs-10/11

Catalog number: 18129

Epoch time: 88031.89723904
Element set: 256
Inclination: 82.9230 deg
RA of node: 249.1248 deg
Eccentricity: 0.0012177
Arg of perigee: 11.5921 deg
Mean anomaly: 348.5502 deg
Mean motion: 13.71887422 rev/day
Decay rate: 1.3e-07 rev/day²
Epoch rev: 3052

The Propulsion Systems of the Phase-III Series Satellites

by Richard L. Daniels, W4PUJ
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(Presented at the 1st. Utah State University
Conference on Small Satellites, October 7, 1987)

ABSTRACT

The radio amateur satellite community pioneered in the design, fabrication and launch of small, low-cost satellites. While these satellites became increasingly more sophisticated, they continued for years as piggyback passengers with no capability for orbital adjustment once released. Beginning with the Phase III spacecraft series an onboard propulsion capability was included permitting significant adjustments to the initial orbit. This paper describes the design of the propulsion modules of three Phase III spacecraft initially using a solid rocket motor and moving to a bi-propellant liquid propulsion system in following spacecraft. While the specific hardware and system design utilized may not be specifically applicable, the general approaches taken could point the way to satisfying the propulsion requirements of other small, low-cost satellites.

1.0. Introduction

On January 12, 1961, a small, simple telemetry beacon satellite was carried into orbit as a piggyback passenger on the Discoverer XXII launch. Battery powered, this satellite operated for only a few days sending crude telemetry signals to amateur radio operators around the world. The product of a group of California radio amateurs organized as Project OSCAR, this satellite was named OSCAR I, the acronym standing for, Orbiting Satellite Carrying Amateur Radio.

In the ensuing 25 years the amateur satellite community has expanded internationally and the products of their labors have evolved into long-lived, sophisticated, communications satellites operated as free-access relays for amateur radio operators around the world. Since 1969 the Amateur Radio Satellite Corporation (AMSAT), an East Coast successor to Project OSCAR, along with its international affiliate organizations, has provided the focus for amateur radio satellite activity in the western countries. Beginning in 1978, a similar series of amateur radio satellites was initiated by amateur radio enthusiasts in the USSR.

Having demonstrated a capability for building and operating long-lived, multi-transponder communications satellites in low earth orbit, AMSAT groups in the U.S. and West Germany undertook the development of what was seen as the next step in the program, the Phase III series of spacecraft. This terminology derived from the categorization of the early, short-lived, beacon satellites as the Phase I program and the following long-lived

communications satellites in low earth orbit as the Phase II program. The Phase III effort was directed to designing a sophisticated spacecraft with multiple transponders to be placed in a highly inclined elliptical orbit similar to one pioneered by the USSR Molniya satellites.

The Molniya-type orbit would permit much greater communications coverage from its apogee of 35,000 Km and would extend visibility time for individual ground stations from the 20 minute average for the low orbiting satellites to as many as 10 hours at a stretch. In order to achieve this orbit, however, it would be necessary to bring into the design one capability never before incorporated into amateur radio satellites, on-board propulsion. The only available launch opportunities were for piggyback rides with commercial communications satellites on their way to geosynchronous orbits. The plan developed was to ride with the primary payload into a low-inclined geosynchronous transfer orbit and then use an on-board propulsion system to change the inclination of the orbital plane to 63 degrees. As part of the same operation, perigee would be raised a safe height while apogee was to be left at the 35,000 Km of the transfer orbit. The following paper describes the development of three propulsion systems to accomplish this goal.

2.0 The Phase III Spacecraft

The Phase III spacecraft is shown in Figure 1 and an exploded view of the second mission spacecraft in Figure 2. This distinctive three-armed structure, the shape of which was initially driven by space availability on an early launch possibility, has proven to be an excellent design both structurally and in optimizing the illumination of the six solar arrays. The cylindrical center section was sized to house the spherical solid rocket motor that was available for the first mission. The spacecraft was spin stabilized with the spin axis coincident with the center line of the central cylinder.

Attitude control with accurate pointing capability was essential both during the motor burn, and later for pointing the high-gain antennas toward earth at apogee. This was provided by an attitude control system utilizing magnetic torquing with earth and sun sensors for reference. An on-board computer, the Integrated Housekeeping Unit (IHU), had among its principle responsibilities the taking of data from the sensors, comparing the readings with three navigational reference systems resident in memory, determining the current

spacecraft attitude, and activating the torquing magnets contained in the three spacecraft arms to move the spin axis to the desired orientation. While this system did not permit rapid change, experience has shown that it can effectively spin up the spacecraft from essentially no

rotation to the 50 RPM necessary for motor burn and point the spacecraft with the necessary precision.

In addition to controlling spin rate and attitude control, the IHU is responsible for overseeing a wide range of housekeeping duties that include: sequencing of the operating schedules for the communications transponders; storing ground commands for later implementation; controlling the charge rate from the solar arrays so as to keep the nickel-cadmium storage batteries at the proper charge level; and, most importantly from the propulsion standpoint, receive, verify, store, and issue motor firing commands at the specified time and point to the proper direction in space.

3.0 The Phase IIIA Propulsion System

The first Phase III propulsion system was designed around use of an available spherical solid rocket motor. Design, fabrication, handling, control, safety and operational results are addressed in the following.

3.1 The Motor

The Phase IIIA propulsion system was designed around the Thiokol TE-345, a 13 inch spherical solid rocket motor. This motor, originally designed as retro rocket for the Gemini spacecraft was now being used in other space programs and arrangements were made for the donation of one motor to AMSAT. The total impulse available was calculated to be just sufficient to achieve the desired Phase III final orbit. Mounting required only providing the required attachment interfaces in the spacecraft structure and careful alignment of the motor nozzle with the spin axis.

3.2 The Motor Ignition Unit (MIU)

The electronics to initiate the ignition sequence were contained in the MIU module. The electronics in this box were designed to control the ignition sequence and to generate the high-current, low-voltage pulse necessary to assure igniter performance. The cabling associated with the MIU included a heavy-duty normally open relay in the igniter firing circuit, interconnection with the IHU computer, and a safe/arm plug that provided complete isolation of firing circuits prior to launch. After launch and determination of the initial orbit parameters, the spacecraft was spun up using the torquing magnets, and a series of firing commands were issued to the spacecraft. These included an enable code, a firing code, and the precise time for motor firing.

These commands were stored in memory and at the appropriate time the ignition sequence was initiated by the IHU issuing the enable and firing codes in the proper sequence and timing. The MIU first compared the received codes against matching codes stored in its electronics, and then initiated the firing sequence. This involved: first, closing the relay that had isolated the igniter circuits; then, energizing the switching regulator in the MIU to generate the firing pulse. Any deviation from the planned timing or sequence of events triggered an immediate shut-down command.

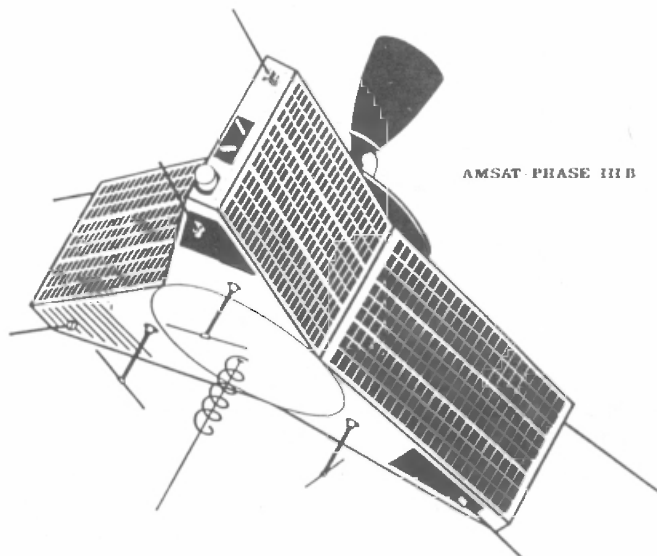


Figure 1: AMSAT Phase III-B Satellite.

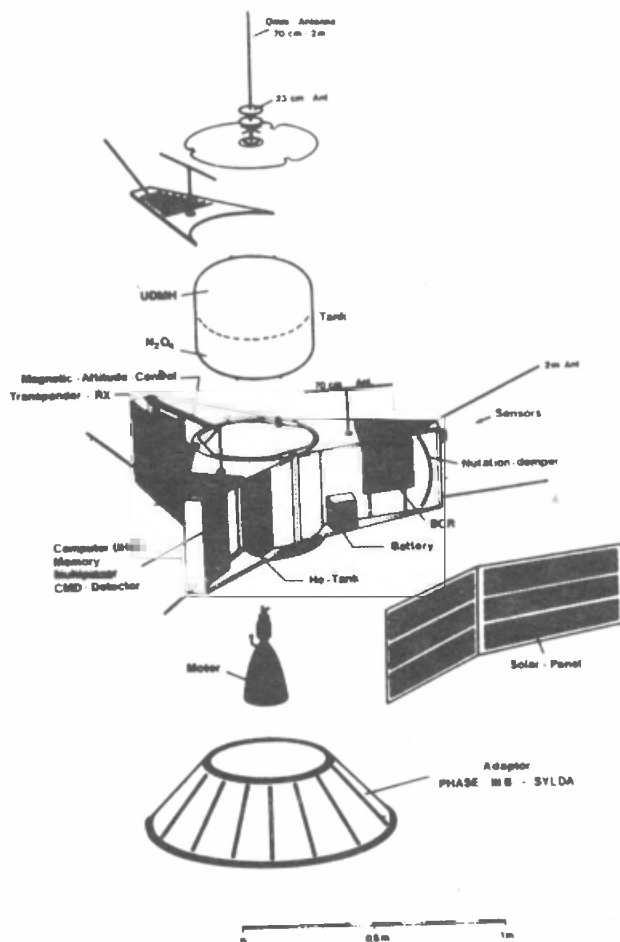


Figure 2: Exploded View, Phase III-B Satellite.

3.3 Fabrication

With the exception of the motor and its two igniters, all other components utilized in the Phase IIIA propulsion system were readily available from commercial sources consisting largely of cabling, connectors and small electronic components. The principal consideration was assuring absolute reliability of the design and the components used in the system to assure that the motor ignition would occur when commanded and not at any time before.

3.4 Safety Considerations

Any system that incorporates almost 40 kg of explosive material must be considered a hazardous device requiring careful handling and detailed safety planning. On the other hand, one advantage of using a solid rocket motor was the relative simplicity of handling from a safety standpoint. The motors currently in use have evolved to the point that they are relatively immune to ignition by static electricity discharge or low-level stray currents. The Motor Ignition Unit and associated cabling was carefully designed to assure the isolation of all firing circuits by open relay points, safe/arm plug interrupts and an absolute requirement for recognition of properly coded enable and firing keys.

Hazardous operations were delayed until as late as possible in the launch campaign. However, beginning with the installation of igniters in the motor, and continuing with all subsequent activities, hazardous conditions were assumed and extreme care was required to be exercised. The motor, spacecraft and handling personnel were all required to be connected to a common ground point. Any time that new structural components were brought together, they too were brought to the common ground. Equally important, in integrating the spacecraft with the launcher was the need to electrically check all cabling that could carry firing current to assure proper termination and absence of any voltage prior to mating connectors. The last action before final close-out for launch was to replace the "Safe" plug with the "Arm" plug to establish connections between the MIU, the spacecraft battery and the firing relay. This established all necessary wiring connections for motor ignition but still absent were the firing keys which are only communicated to the spacecraft after launch.

After detailed verification and extensive testing, the system here described proved to operate in a consistently reliable manner. The importance of using reliable components and a solid system design cannot be overemphasized for this type of system. Particular attention should be given to identifying any individual components that could, through failure, initiate a firing sequence. Further, the adequacy of the design and operating procedures must be reviewed by and acceptable to the safety staff of the launch authority. Early interaction with the safety people is recommended to assure a complete understanding of the requirements for safety and to provide enough time to take any corrective action that may be necessary.

3.5 Operations

The proper operation of the Phase IIIA propulsion system, to the extent it could be tested on the ground, was completely verified through the testing program. The safety features of the design and ground handling procedures were reviewed and accepted by the safety staff of the Ariane launch authority. On May 23, 1981 final launch preparations at the Kourou launch site went smoothly without any major hitches. Shortly after lift-off, however, failure of one of the four first stage engines resulted in a catastrophic failure of the launcher and payloads. As a result, there was no opportunity to demonstrate the operation of the Phase IIIA propulsion system.

4.0 The Phase IIIB Propulsion System

Following loss of the Phase IIIA spacecraft, the design and construction of a replacement spacecraft was immediately undertaken. While the basic configuration and design of the Phase IIIA was retained, a number of improvements were undertaken for the communications modules. Beyond these, the most significant change was in the propulsion system. With improvements adding weight to the spacecraft and a launch opportunity that would provide a lower initial orbit inclination, it was clear that the solid motor could not provide enough impulse to assure reaching the desired orbit. The decision was made to explore other approaches.

A breakthrough was made when the German team, AMSAT-DL, was able to make arrangements with the German aerospace firm, Messerschmitt-Boelkow-Blohm, for donation of a 400 N bi-propellant motor with associated hardware and ground handling support. The availability of this hardware made possible a move to a liquid propellant system. Calculations clearly indicated that a significant improvement in total impulse could be achieved and that the capability for multiple motor burns added significant mission flexibility. On the other side of the ledger, the propulsion system would be more complex and potentially hazardous than the solid motor system used in the previous design.

The configuration of the phase IIIB spacecraft is shown in Figure 2 and the propulsion system configuration in Figure 3. The redesign utilized the total volume of the central cylinder for a two chambered propellant tank that was pressure fed by a high pressure helium bottle through a plumbing system designated the Propellant Flow Assembly (PFA). The propellants being hypergolic, ignition was achieved simply by opening the motor valves. Following is a discussion of the design and system components.

4.1 The 400 N Thruster

The motor provided by MBB had originally been developed as a vernier engine for the Europa European launch vehicle and was designed to give about 400 Newtons (95 Lb.) of thrust. The original propellants for this motor were the Nitrogen Tetroxide (N2O4) and

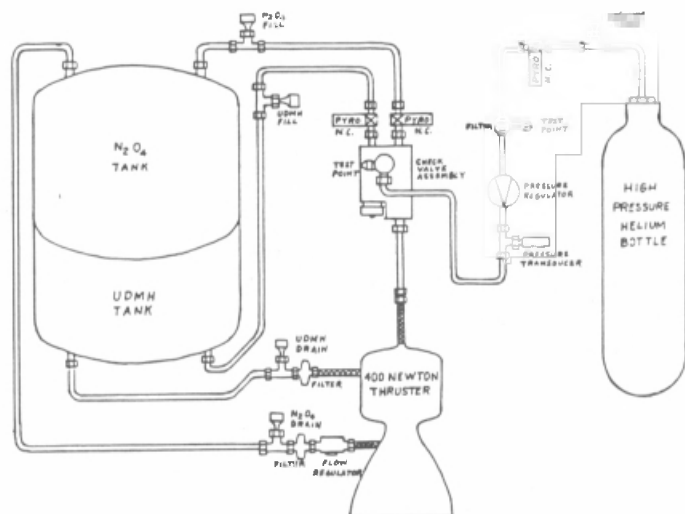


Figure 3: Phase III-B Propulsion System.

Aerzine 50 (AZ-50) propellants utilized by the Europa first stage. Following cancellation of the Europa project, the motor was selected as the apogee motor for the Symphonie joint German/French communications satellite project with modification for use of Monomethyl Hydrazine (MMH) instead of AZ-50. In discussion with the MBB engineering staff it was decided to further modify the AMSAT engine to permit use of Unsymmetrical Dimethyl Hydrazine (UDMH) instead of AZ-50 because this propellant was more readily available at the launch site.

4.2 Material Compatibility

The propellants used raised major problems of material compatibility. N2O4 is not only extremely active and will readily attack most organic material, it is also easily contaminated. UDMH, while not as corrosive, is equally subject to contamination. Both are extremely toxic and cannot be handled without use of protective suits and air filtering. To compound the problem, the close tolerance on valves and pressure regulators meant that both the gas and fluid flow transport systems were extremely vulnerable to contamination. Although MBB had agreed to provide certain key hardware items such as the pressure regulator, explosive valves, and a multiple check valve assembly, the remaining components had to be procured and the propulsion system designed and assembled by the AMSAT group. Materials that were to come in contact with the propellants were restricted to selected types of stainless steel and aluminum alloys with teflon for seal material.

4.3 The Propellant Tank

Initially it was planned to fabricate the two-chambered propellant tank from a special alloy of stainless steel. When it was found that this material could not be formed properly, the decision was made to go to an aluminum tank design. The most important requirement was to provide an intermediate bulkhead that would give absolute assurance that the two propellants could not come into contact through cracks or poorly welded seams. The

design that achieved this is shown in Figure 4. The tank was constructed in three sections milled from thick billets of aluminum alloy so that the intermediate bulkhead was an integral part of the center section and the welded seams were isolated to the individual propellant tanks. Thus, any seam was avoided that upon failure would result in propellant mixing. Drain points were located to keep the tube mouths covered by propellants through the effects of the spacecraft spin rate and the effects of propulsion thrust. The tank took advantage of the full volume of the central cylinder with only enough space at the top and bottom for tubing and cable access.

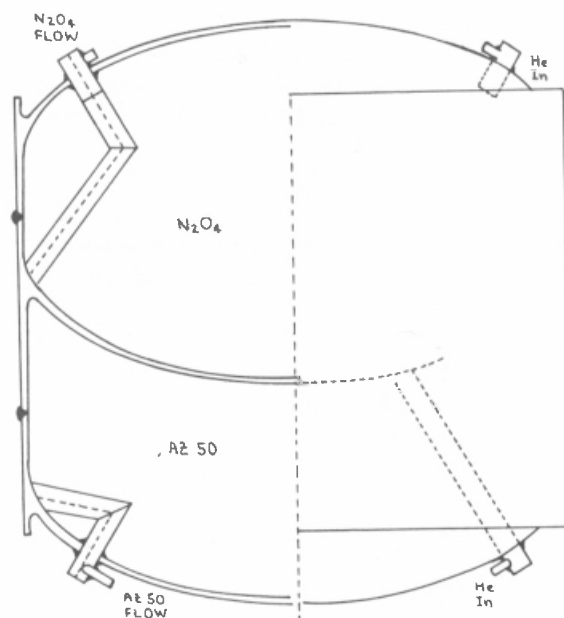


Figure 4: Propellant Tank.

4.4 The Helium Bottle

The flow of propellants to the motor was forced by helium pressure. In order to provide sufficient helium gas to assure total displacement of the propellants it was necessary to store the helium at a very high pressure. Since a commercial bottle was not available in a usable configuration and at an affordable cost, a low cost, but effective alternative was developed. This is shown in Figure 5. The AMSAT-DL group located a metal bottle that was inexpensive and commercially produced as a fire extinguisher bottle. The problem was that this bottle was rated to a pressure well below the helium storage bottle requirement of 400 atmospheres (6,000 psi.). The solution was to encase the length of the bottle with carbon/epoxy fiber windings, thereby increasing the burst pressure of the bottle to a demonstrated 1132 atmospheres.

4.5 The Liquid Ignition Unit (LIU)

The electronics module for initiating and controlling the motor burn for the Phase IIIB spacecraft was designated the LIU to differentiate it from the Phase IIIA MIU. Its purpose was similar to that of the MIU but with one important additional requirement. This unit was now also required to control the burn time of the

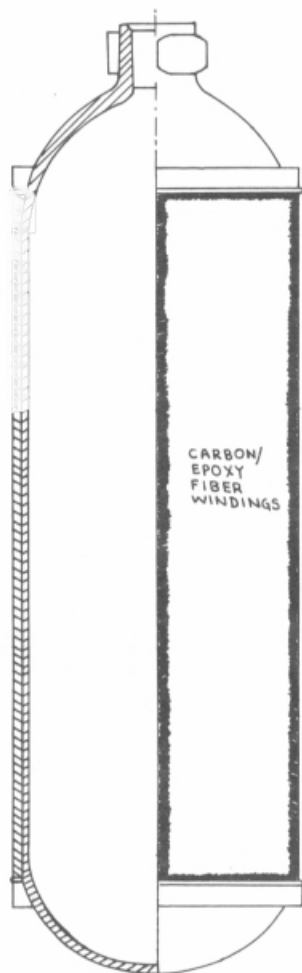


Figure 5: Helium Bottle with Reinforcement Windings.

motor. To do this the LIU design included a counting capability in 50 millisecond intervals to meter the duration of an individual burn.

4.6 Propellant Flow Assembly (PFA)

With the exception of the electronics package, tankage, and interconnecting tubing, the remaining components of the propulsion system were assembled on a single mounting plate as shown in Figure 6. As can be seen from the drawing, the PFA was a consolidated module containing all fill valves, pressure regulation, filtering and check valves to prevent backflow in the pressurization lines. It also included three normally closed explosive valves to isolate the high pressure helium tank and the two sections of propellant tank until the initial firing sequence was commanded in orbit. Also included in the PFA design was a precision pressure transducer to measure the low pressure side of the system and transmit its measurements by telemetry to the ground. A much more coarse measurement of the high pressure helium bottle was made using a strain gage attached to the wall of the bottle.

In designing the PFA every effort was made to minimize the number of wrench-tightened fittings and to use welded connections wherever possible to reduce the possibility of leaks. In addition, all subassemblies were pressure and leak tested when finally assembled.

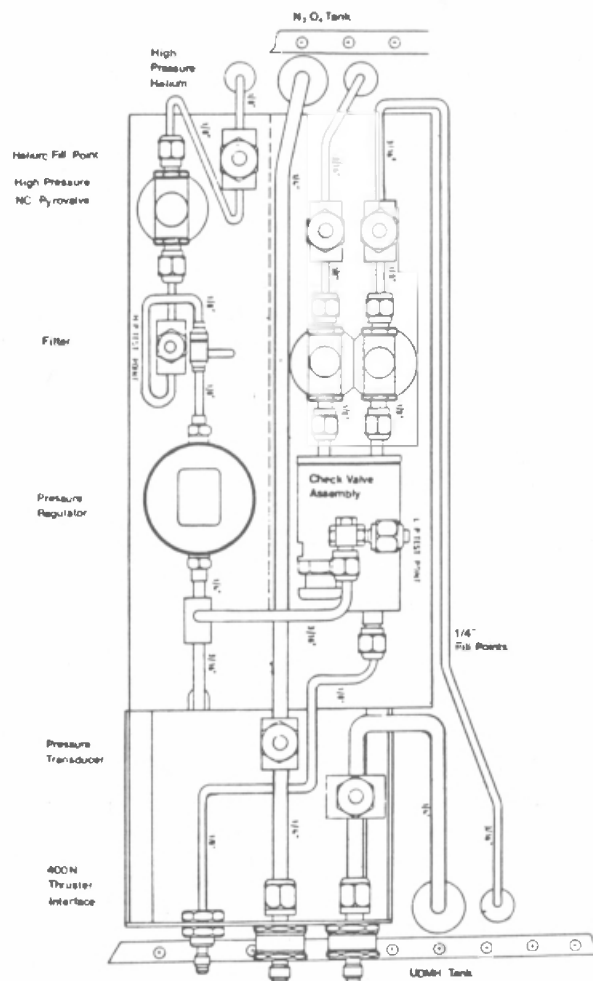


Figure 6: Phase III-B Propellant Flow Assembly.

The ignition sequence, following receipt of validated keys and commands from the IHU, involved actuating the explosive valves in the proper sequence, allowing the system to stabilize, and then opening the motor propellant valves. Valve operation in the 400N thruster is initiated electrically but activated by helium pressure. Following ignition, the LIU counted the commanded time intervals and then initiated a shutdown sequence. The Phase IIIB mission profile included two major burns to achieve the desired orbit with enough residual fuel for trimming maneuvers, if required.

4.7 Safety Consideration

All of the safety considerations addressed for the Phase IIIA spacecraft apply to the Phase IIIB system. To these was added the problem of safety handling very toxic, self-igniting propellants. This required even more detailed and careful planning for ground handling at the launch site including hazardous propellant transport, complex filling operations, and subsequent careful ground handling of the loaded spacecraft. A close interaction with the launch authority safety staff before arrival at the launch site and during the launch campaign was essential. In addition, arrangements had to be made with the launch authority for provision of protective suits and transport of propellants. Once fueled, the spacecraft required constant monitoring to assure that there were

no hazards to personnel from release of toxic vapors. Finally, there was the problem of cleaning the ground support equipment used to fill the spacecraft after the tanking operation was complete. A new safety feature initiated with the Phase IIIB mission was a CRT safety display driven by the spacecraft computer that was located at the range safety officer's station showing in green and red the status of the firing keys in memory, which plug was in the safe/arm connector, what parts of the system were powered up, and how much time had elapsed since the last telemetry update.

4.8 Operations

The Phase IIIB spacecraft and its propulsion system were successfully launched on an Araine rocket from the Kourou launch site on June 16, 1983. Initial orbital operations were disrupted by an unplanned collision between the spacecraft and the third stage that resulted in damage to an antenna, change in the spacecraft orientation and reduction in the spin rate. Despite this initial difficulty, the spacecraft was reoriented, spun up to the proper rate using the magnetic torquing system and the propulsion system commanded into its first burn. Telemetry following the successful burn indicated some deviation from the expected burn time, but otherwise the operation of the system appeared nominal. The orbit actually achieved with the first burn was, as planned, intermediate to the one finally desired. Action then was initiated by the ground stations to reorient the spacecraft in preparation for the second burn. During this somewhat lengthy process, telemetry indicated a steady drop in helium pressure.

By the time the spacecraft was reoriented for the second burn, the helium pressure had fallen below the level necessary to actuate the motor valves. Thus, the intermediate orbit became the final orbit for this mission. The problem is thought to have been caused by a leak in the seal of the high pressure helium bottle triggered by the rapid colling of the bottle during the first burn. The orbit achieved proved to be very useful if not as effective as the one desired. The Phase IIIB spacecraft, renamed OSCAR 10 once in orbit, achieved most of its mission objectives. It provided effective communications for the world amateur radio community for three years. It is only now nearing the end of its operational life due to damage to the computer memory from the effects of radiation.

5.0 The Phase IIIC Propulsion System

At the time the decision was made to undertake the PHASE IIIB development, a policy was established that two of each of the electronic modules would be produced. This was done to provide redundancy in case of last minute problems with the prime modules and to provide the key hardware for a follow-on mission. Not long after OSCAR 10 began orbital operations, the development of the Phase IIIC spacecraft was initiated. As with the Phase IIIB system, improvements and augmentations were made in the design of the communications sub-

systems. For the propulsion system, it was decided to continue with the 400N thruster. MBB agreed to make available another motor, but was not able to provide the supporting hardware beyond a minimum number of fill and drain valves since supplies had been exhausted in supporting the previous mission. The challenge, therefore, in developing the Phase IIIC propulsion system was to come up with a redesign of the Propellant Flow Assembly that used available and affordable hardware.

The results of this redesign effort are shown in Figure 7. With the exception of the 400N thruster and three specialized fill and drain valves, the complete system was designed using commercially available and relatively inexpensive hardware. The design is an absolutely bare bones approach emphasizing simplicity and economy. The propellant and helium tanks were used without change except for a modification in the propellant tank to accommodate use of AZ-50 rather than UDMH as the fuel and modifications to the helium tank to preclude a repeat of the leak experienced with the PHASE IIIB system. The major changes were in the PFA.

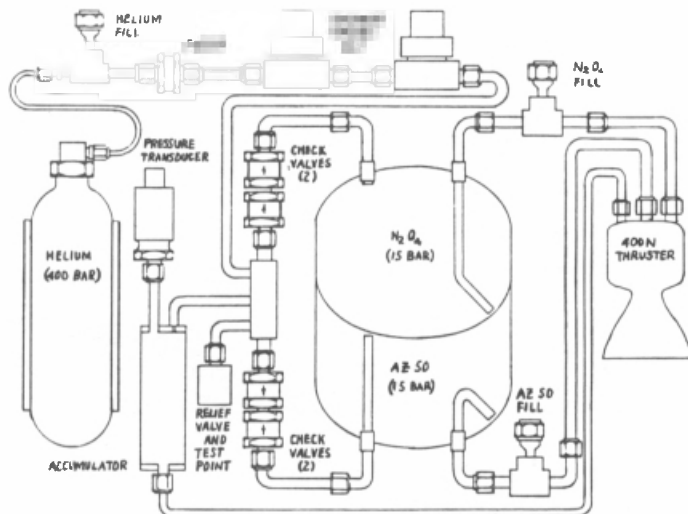


Figure 7: Phase III-C Propulsion System.

5.1 The New Propellant Flow Assembly

As shown in the schematic in Figure 7 and the drawing in Figure 8, significant changes were made in the design of the PFA. First, the normally closed explosive valve that had isolated the high pressure helium bottle was replaced by two high quality, low leakage, electrically operated valves. Further, when the system is actuated, one of the valves is held open to initiate the flow of helium, the other is operated in a feedback loop with the pressure transducer to provide pressure regulation in a "bang-bang" mode. In this operation, the valve is opened when pressure falls below a set level and closed when the desired pressure is reached. Should the pressure rise above the expected level, the role of the valves is reversed with the suspect regulator valve now held open and the other operated in the regulating mode.

A further safety feature was provided by adding a relief valve set to operate if the propellant tank pressure

rises significantly above the operating pressure. Anticipating undesirable pressure fluctuations from the pulsing of the regulating valve, an accumulator was incorporated. The explosive valves and the specialized check valve assembly used in the Phase IIIB design to prevent backflow from the propellant tanks was replaced by redundant commercial check valves. The result of this redesign is a significantly different Propellant Flow Assembly in appearance, but one that is expected to functionally duplicate the operation of its predecessor.

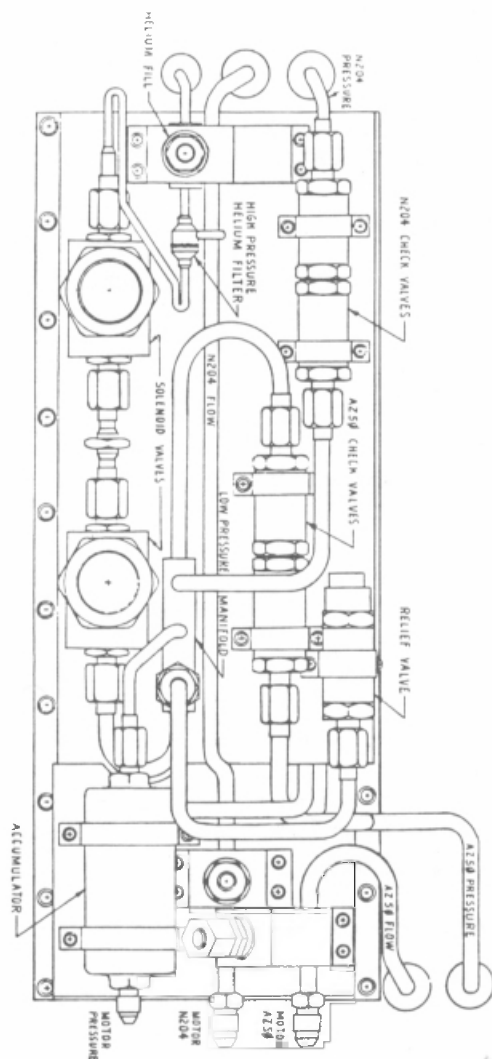


Figure 8: Phase IIIC Propellant Flow Assembly.

4.2 The New Liquid Ignition Unit (LIU)

The redesign of the PFA drove significant redesign of the LIU. While the firing command verification and motor burn timing functions were retained without change, the circuitry required to fire the explosive valves was deleted and new circuitry was incorporated to control the operation of the pressure regulating valve system.

4.3 Current Status

All hardware supporting the new propulsion system has been fabricated, tested at the subsystem level, and assembled into the Phase IIIC spacecraft. Final testing of all spacecraft systems including the propulsion system has been delayed by the schedule slips associated with the Ariane mission V18 failure and recovery activity. Integrated spacecraft testing has now been resumed with a vibration test scheduled for early November. At this time, a total check-out of the propulsion system short of actual ignition will be accomplished. Following the launch that is currently scheduled for early 1987, it is expected that the Phase IIIC propulsion system will function as designed and accomplish the first fully successful orbital maneuver in the Phase III spacecraft series.

REFERENCES

1. J. A. King, "The Third Generation," Part 1, *Orbit*, Vol. 1, no. 3, Sept./Oct. 1980, pp. 12-18; Part 2, *Orbit*, Vol. 1, no. 4, Nov./Dec. 1980, pp. 12-18
2. M. A. Schwende, "Development of Bi-Propellant Orbit Injection and Attitude Control System," Paper No. 78-1092, AIAA/SAE 14th Joint Propulsion Conference, Las Vegas, Nevada, July 25-27, 1978.

Spaceframe Design Considerations for the Phase IV Satellite

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ABSTRACT

Much has been written about the Phase IV mission in a geostationary synchronous orbit, but little has been noted as to what type of spacecraft design can meet these requirements. This paper will show some of the many design tradeoffs needed to achieve an initial spacecraft configuration to meet the requirements.

Introduction

A very thorough discussion has been previously presented on the current status of communications through the use of amateur satellites.¹ This discussion also laid the ground work for the need for a transponding satellite, in geostationary synchronous orbit, providing continuous communications links unhampered by the frailties of ionospheric propagation. Information was also presented showing the requirements of the communications transponders and the total link budget.

Not previously discussed was just what type of satellite is required to provide the platform for these transponders, how much solar power will be required to fulfill the needs of the mission, and how do we squeeze the needed design into a shape that will permit us to be secondary payload (read that as lowered launch costs) rather than be so large as to be deemed the primary payload with its unreachable costs.

There are only a few feasible methods of designing a satellite for this type of mission and to be able to simultaneously solve several conflicting problems dealing with antennas, attitude control methods and thermal design. Not uncommonly, solutions to some of these problems may seem like compromises. The configuration that AMSAT has achieved, to date, is not necessarily new, but it has not been too widely practiced in the sphere of geostationary satellites. Our attitude control methods are even a bit revolutionary, and heretofore unflown. Venturing into the unproven world of spacecraft design is not new for AMSAT, and so we feel comfortable basing our design on sound, albeit new engineering methods.

Mission Design

Detailed requirements of a Phase IV mission spacecraft have been initially defined in considerable detail and will not be examined in depth here.² The general precepts of this mission can be outlined as follows:

- 1) A geostationary Earth synchronous satellite placed to eliminate antenna re-orientation and tracking efforts by ground stations.
- 2) Continuous, 24 hour per day availability.
- 3) Multiple transponders with Mode J/L and Mode S capabilities suitable to serve a broad range of Amateur communications needs.
- 4) Transponder power outputs of 80 and 120 W PEP, depending upon the band.
- 5) Directive antennas with gains in the range of 16 dBi on all bands.
- 6) Solar power generation rating of at least 235 Watts.
- 7) Propulsion capabilities to place the spacecraft into the geosynchronous orbit from a geosynchronous transfer orbit (GTO), delta V in the range of 1500 m/s.
- 8) Station keeping propellants.
- 9) Attitude control system needed to maintain antenna orientations.

All other features of the spacecraft are subservient to these prime requirements. Many compromises and innovations must be made to achieve these goals. Further, there is more than one spacecraft configuration that can satisfy these needs, the trick then is to find the configuration that will meet these needs and also be producible in an AMSAT environment.

Basic Geostationary Spacecraft Configurations

At the altitudes for a geostationary spacecraft, the Earth presents problems, as a trade-off for achieving synchronous positioning. These problems take the form of a relatively small target of about 16 degrees; thus the satellite antennas need to have gain and directivity to concentrate the RF energy on the target Earth and avoid wasted RF energy needlessly irradiating unwanted "empty" space. This then means that the antennas need to be carefully pointed toward Earth at all times, in deference to the large signals from close-by Low Earth Orbit (LEO) satellites.

The Earth also does not provide any significant impact to the heating of a spacecraft at the 36,000 km altitudes typical of geostationary satellites. The only heating influence, other than internal power dissipation, is that of the sun. (See Figure 1.) Since the side walls of most spacecraft are covered with solar cells and need to absorb solar energy to function, that same absorption will cause excessive solar cell temperatures, unless

special precautions are taken. Solar cells exposed to direct sunlight without any special provisions for cooling can reach temperatures of up to 119°C, seriously degrading the cell's ability to produce electrical energy. Truly useful solar cell temperatures are below 40°C, hence a thermal design paradox for an oriented antenna array spacecraft.

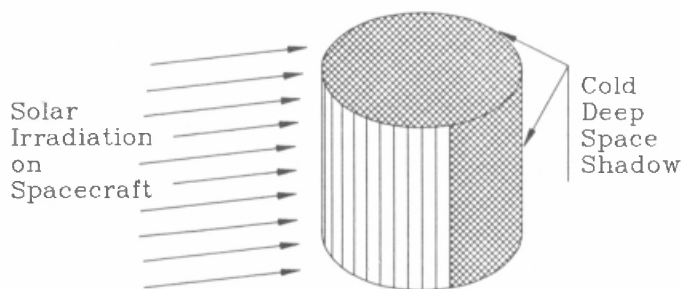


Figure 1. Spacecraft Solar Heating

Geostationary Spacecraft Designs

One solution to the solar cell heating dilemma is to spin the satellite, rotisserie fashion, so as to limit the solar exposure time of any particular cell, allowing it to cool on the shadowed side of the spacecraft. This spinner concept is shown in Figure 2, but the concept is not without its problems. To maintain the communications antennas oriented on the target Earth, they must be "de-spun" from the rotating spacecraft body, a very complex mechanical and electrical problem requiring bearings, motor driver and control, many electrical sliprings passing either significant power or RF energy and a precision pointing control system to maintain the antenna orientation. While this method provides solutions to the severe thermal heating of the solar cells, the induced problems of despinning the antennas become very expensive to implement.

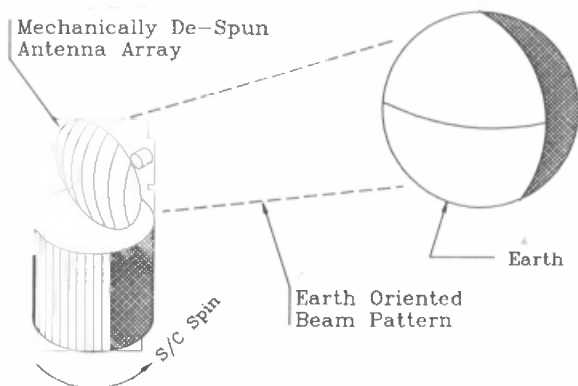


Figure 2. Spin Stabilized Geosynchronous Satellite

A second solution to this problem also employs the spinning of the satellite, but the antennas are electrically de-spun. (Figure 3) While this method alleviates the mechanics of the de-spun antenna, just mentioned, the burden is now placed upon a complex RF phasing/switching scheme, that may be difficult to implement with the antenna gains that we desire in the frequency ranges that we intend to use for such a mission.

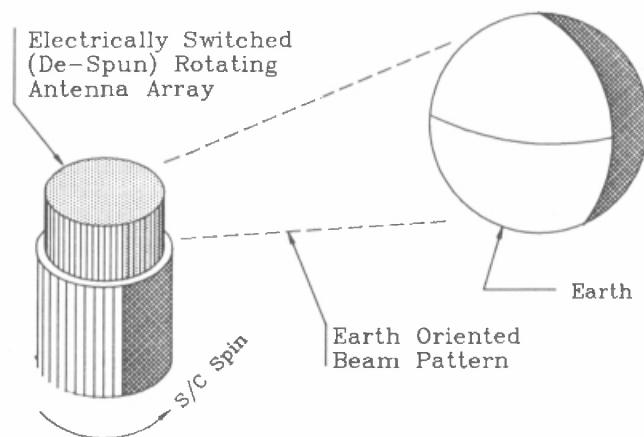


Figure 3. Spin Stabilized Geosynchronous Satellite

Clearly, a fixed pointed array of VHF, UHF and microwave high gain antennas is needed for the success of Phase IV. We need as simple a spacecraft as we can possibly devise. (Figure 4) Rather than complicate the mechanics or electronics of a spacecraft to solve the high temperature solar cell problem, we needed to answer the thermal problems with a thermal device. Reasoning that if a method could be found to transport the excess absorbed solar energy from the sun side of the spacecraft to the shadowed side, we would solve two problems at once, cooling the hot side and warming the shadowed side. Such a solution will also set the basic bounds for the thermal environment of the spacecraft interior.

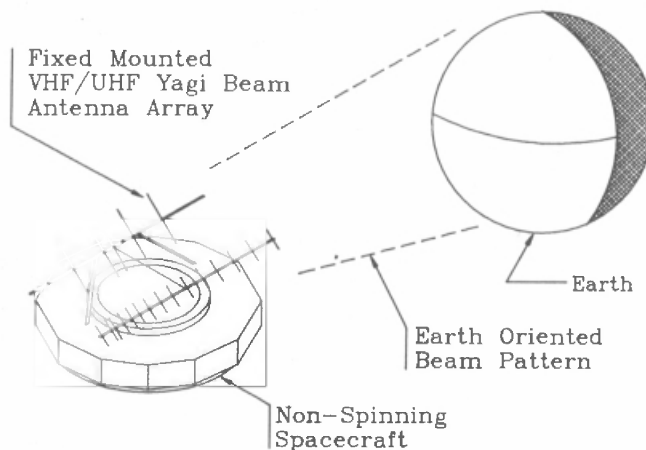


Figure 4. Body Stabilized Phase IV Geosynchronous Satellite

Phase IV Design

To satisfy the essential power generation needs of the Phase IV spacecraft, a considerable area of solar cells will be needed to be exposed to the sun. Mounting these cells onto the skin of the spacecraft for at least a 90% coverage (Phase III spacecraft only had 70% area coverages) and using standard high output silicon cell technology with 12.5% conversion efficiency, the 235 W output can be met with a projected area of 1.50 m². This, roughly is the product of the diameter and length of the spacecraft and the actual area of cells on a cylindrical shell design would be $1.50 \cdot \pi = 4.7 \text{ m}^2$.

The 1.50 m² requirement can be met by a range of diameters and lengths of design; but remember that we need to be short and squat to fit into a launcher as a secondary payload. If the length were to be 0.65 m, then a diameter of 2.30 m would meet the basic requirement. The importance of this 2.30 m dimension will be shown later.

Solar Panel Cooling with Heat Pipes

Computations of the heat flow needed to cool the sun-side solar cells show that the total amount of heat to be transported is not large. It is the distance of that transport, and the relatively small range of temperatures available, that required a very high value thermal conductor or other heat transfer system. More than twenty years ago G.M. Grover independently conceived such a device in the idea of the heat pipe which had its antecedents in devices first advanced by R.S. Gaugler in 1942. Grover also coined the name of "heat pipe" which is shown diagrammatically in Figure 5.

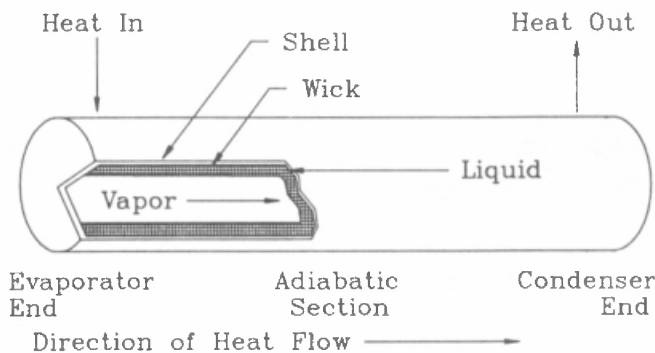


Figure 5. Basic Heat Pipe

Design reference literature provides some useful description of a heat pipe.³ "The heat pipe is a thermal linkage of very high conductivity. It is a closed, evacuated chamber lined with a wick. Heat is transported by evaporation of a volatile fluid, which is condensed at the cold end of the pipe and returned by capillary action to the hot end. The vapor passes through the cavity. Heat pipes consist of three zones or sections: the evaporator, the condenser and an adiabatic section connecting the other two. In some designs, the adiabatic section may be very short. This device offers a number of important properties useful in electronic equipment cooling systems. It has many times the heat transfer capacity of the best heat-conducting materials while maintaining an essentially uniform temperature and transporting heat over distances of several feet. It requires no power and operates satisfactorily in a zero-gravity environment."

A substantial engineering study has been conducted to determine the limits of applicability of heat pipes to the Phase IV thermal design. That study has resulted in a level of confidence of performance and design information that now allows us to proceed with this design precept. While the use of a heat pipe design is not without its penalties, specifically a total solar panel and

heat pipe sub-system mass of about 51 kg. Measured against some of the complications of despun antenna platforms, the heat pipe approach penalties seem to be a lesser evil as there are no moving parts or complex RF switching.

The heat pipe engineering study clearly showed the importance of one statement from the noted literature:

"It is very important to realize that, while a heat pipe is intrinsically an excellent thermal conductor, its effectiveness in a cooling application is strictly limited by the thermal resistances at its interfaces with the heat source and sink."

Figure 6 illustrates the solar-panel/heat-pipe design configuration that has been achieved to meet the basic precepts. The heat pipe is an aluminum extrusion with finely detailed internal fin passages to provide the capillary action for a liquid return path. One of the more prominent features of this heat pipe is the integral flange, permitting ready thermal coupling in an area of high heat flux. The solar panel itself is fabricated of thin aluminum face sheeting over a honeycomb core stock, a common design practice. What is not commonly practiced is the use of the aluminum bars integrally laminated into the panel. The placement of these bars permit the use of numerous mounting screws through the heat pipe flange threaded into the bars. These fasteners do not occupy valuable panel area "real estate" on the sun side of the panels and, in combination with the stiffness of the bars, also insure an intimate thermal contact of the solar panels with the heat pipe. This thermal contact between the solar panel and the heat pipe represents the single most critical thermal resistance in this sub-system. Additionally, the contact resistance is also the most uncertain, as it involves such nebulous terms as panel flatness, surface finish, number of fasteners, fastener tightness and the amount of panel distortion induced by the fasteners. Earthly solutions for interstitial resistances, such as found here, often involve fillers or other materials that are completely unacceptable in the space environment.

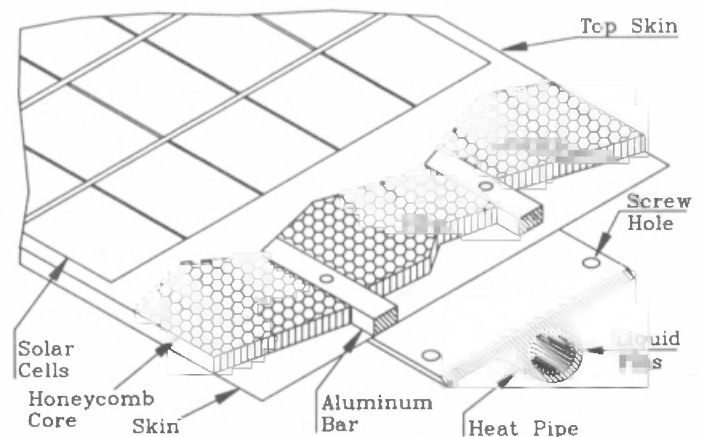


Figure 6. Solar Panel & Heat Pipe

Solar Panel Size Selection

As the selected heat pipe is a commercially produced extrusion, it is typically produced in lengths of only 12

feet. Remembering the reference to the spacecraft diameter of about 2.3 m, the circumference of such a circle is approximately 24 ft. or just two lengths of 12 ft. heat pipe. It is not feasible to join the ends of two heat pipe extrusions of this type and achieve a single passage, as is desired. Each 12 ft. section of heat pipe will need to be a sealed assembly on its own, and we will need to thoroughly join, mechanically and thermally, the abutting ends of the two heat pipes so that they will perform nearly as if they were a singular unit. Were we to have a spacecraft diameter greater than the 2.3 m, then we would need more than two basic sealed heat pipes per ring, at a substantial increase in cost.

Silicon solar cells are not readily attached to curved surfaces so the squat "cylinder" now becomes a polyhedron. A 24 ft. heat pipe ring can be conveniently divided into 12 lengths of 600 mm, so our polyhedron will actually become a dodecahedron. This selection of twelve sides of 600 mm length was more than casually determined. Studies were made of how many 20x40 mm solar cells were required to make up a single series string of cells for the power system. In Phase III that number happened to be 68 cells. By slight readjustment of the battery charge regulator (BCR) circuitry, that string length could be made almost any convenient number, such as the 72 cells that fit onto the current design of Phase IV panel. Having each solar panel an integral voltage unit means that all panels need only to be placed in parallel with an appropriate series blocking diode on each panel. Allowing for manufacturing, assembly and handling needs, individual solar panels of 110 x 600 mm were found to meet all of the competing technological needs. Six such panels on each facet of the twelve-sided spacecraft, a total of 72 identical panels, comprise the total solar array.

The interrogating individual might question the wisdom of dividing each facet of the spacecraft into six solar panels. Indeed, the structural integrity of the spacecraft would be much better if the panels on each facet were singular and they then could act as a stressed-skin panel. Thermal load ratings of available heat pipes, and the lateral thermal conduction properties of the solar panels (getting the heat to the pipes) dictated that six rings of heat pipes be used. As these pipes are individually formed extrusion assemblies, the mechanical assembly tolerances of intimately mounting six heat pipes to a single large solar panel become unfathomable. We are therefore relegated to attacking the structural problems in other manners, while we employ the smaller solar panels individually attached to their respective heat pipe sections.

Solar Panel and Heat Pipe Thermal Performance

Two thermal analytic models have been constructed to initially assess the heat transfer performance of the heat pipe and the resulting temperature profiles of heat pipe and solar panels. Figure 7 shows the dimensions of the basic Phase IV spacecraft bus, as we know it at this time. Figure 8 illustrates the thermal model representation of a single heat pipe ring (one of the six) on the

spacecraft, and Figure 9 is the thermal model of a cross sectional cut of a single solar panel and heat pipe. The values of these two models were cross iterated for a final solution, and in conjunction with several methods of fabricating the individual solar panel. The details of the rejected designs will not be presented, save to say they were less than desirable in performance and fabrication methods.

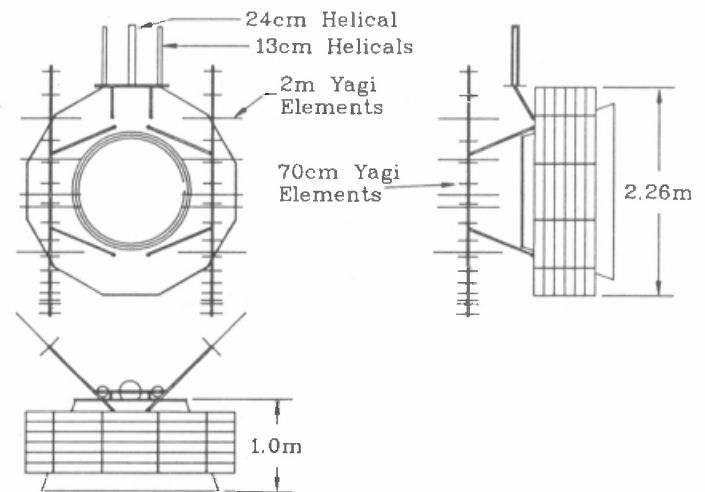


Figure 7. Phase IV Basic Bus Design

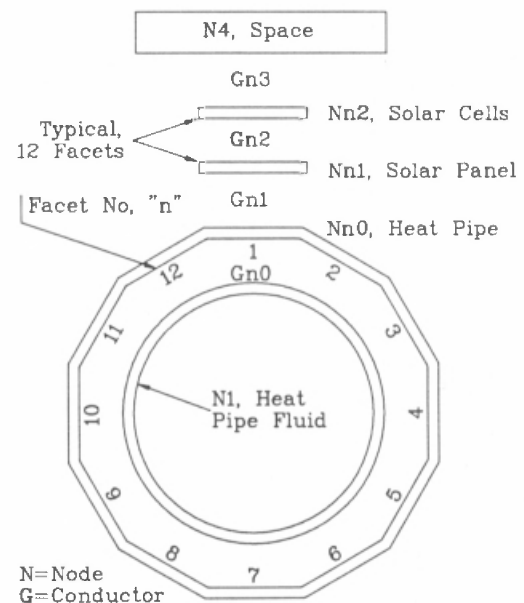


Figure 8. Heat Pipe Ring Thermal Model

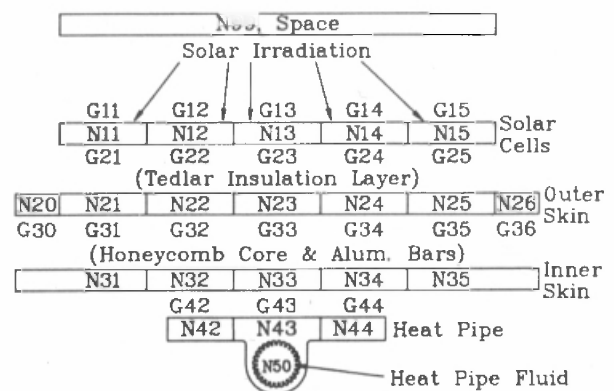


Figure 9. Solar Panel & Heat Pipe Thermal Model

The analyses conducted were run on a Zenith Z-248 computer using a PC descendent of SINDA⁴ (quite a marvel in itself to be able to do on a PC what formerly required a mainframe computer). These results are presented in Tables 1 through 4, showing the temperatures arranged in spatially relatable forms. Tables 1 and 2 show the analytic results for the heat pipe ring with one facet normal to the solar energy, and then rotated 15° to determine the effect of the sun's relative motion around the spacecraft and the varying irradiation of solar panels.

Solar cell electrical output is manifest in a change in

Table 1. AMSAT Phase IV Spacecraft Thermal Analysis
Heat Pipe Ring With Solar Panels Mounted

Solar Cell Absorptance = 0.719
Solar Power Generation = 38.4 W
Azimuth Angle THETA = 0. Deg.
Heat Pipe Fluid Temp. = 5.1°C

Facet No.	Temperatures °C			Heat Flows W	
	Cells	Panel	Pipe	Pipe	Solar
7	-5.0	-1.1	3.9	-16.85	
8	-5.0	-1.1	3.9	-16.85	
9	-5.0	-1.1	3.9	-16.85	
10	-5.0	-1.1	3.9	-16.85	0.00
11	12.2	9.1	5.8	10.44	31.62
12	24.5	16.5	7.2	30.00	54.77
1	29.0	19.1	7.7	37.07	63.24
2	24.5	16.5	7.2	30.00	54.77
3	12.2	9.1	5.8	10.44	31.62
4	-5.0	-1.1	3.9	-16.85	0.00
5	-5.0	-1.1	3.9	-16.85	
6	-5.0	-1.1	3.9	-16.85	

Solar Cell Absorptance = 0.719
Solar Power Generation = 39.8 W
Azimuth Angle THETA = 15.0 Deg.
Heat Pipe Fluid Temp. = 6.8°C

Facet No.	Temperatures °C			Heat Flows W	
	Cells	Panel	Pipe	Pipe	Solar
7	-3.4	0.5	5.6	-17.41	
8	-3.4	0.5	5.6	-17.41	
9	-3.4	0.5	5.6	-17.41	
10	5.5	5.9	6.6	-3.23	16.37
11	20.7	14.9	8.3	20.89	44.72
12	29.3	20.0	9.2	34.56	61.09
1	29.3	20.0	9.2	34.56	61.09
2	20.7	14.9	8.3	20.89	44.72
3	5.5	5.9	6.6	-3.23	16.37
4	-3.4	0.5	5.6	-17.41	0.00
5	-3.4	0.5	5.6	-17.41	
6	-3.4	0.5	5.6	-17.41	

Summary:

Average Heat Pipe Input = 57.21 W
Mean Evaporator + Condenser Length = 1.8 m
Heat Pipe QL Product = 103.0 W.m
Heat Pipe Fluid Mean Temperature = 6.0°C

the total absorbed solar energy that needs to be dissipated by the cell. The optical absorptance does not change from a value of about 0.844 (a Phase III value) which is a ratio of the incident solar energy. Since electrical output is also a ratio of the incident solar energy (12.5%) it is necessary to subtract 0.125 from 0.844 to determine the thermally effective value of absorptance when the cell is generating its maximum output. When the BCR regulation does not require full power from the cells, the operating point of the cell is changed and the effective value of absorptance raises, increasing the cell dissipation. Thus, Table 2 shows the heat pipe ring

Table 2. AMSAT Phase IV Spacecraft Thermal Analysis
Heat Pipe Ring With Solar Panels Mounted

Solar Cell Absorptance = 0.844
Solar Power Generation = 0.0 W
Azimuth Angle THETA = 0. Deg.
Heat Pipe Fluid Temp. = 15.4°C

Facet No.	Temperatures °C			Heat Flows W	
	Cells	Panel	Pipe	Pipe	Solar
7	3.8	8.2	14.1	-19.25	
8	3.8	8.2	14.1	-19.25	
9	3.8	8.2	14.1	-19.25	
10	3.8	8.2	14.1	-19.25	0.00
11	23.6	19.9	16.3	11.99	36.77
12	37.6	28.3	17.8	34.25	63.69
1	42.7	31.3	18.4	42.27	73.55
2	37.6	28.3	17.8	34.25	63.69
3	23.6	19.9	16.3	11.99	36.77
4	3.8	8.2	14.1	-19.25	0.00
5	3.8	8.2	14.1	-19.25	
6	3.8	8.2	14.1	-19.25	

Solar Cell Absorptance = 0.844
Solar Power Generation = 0.0 W
Azimuth Angle THETA = 15. Deg.
Heat Pipe Fluid Temp. = 17.8 oC

Facet No.	Temperatures °C			Heat Flows W	
	Cells	Panel	Pipe	Pipe	Solar
7	5.9	10.5	16.4	-19.88	
8	5.9	10.5	16.4	-19.88	
9	5.9	10.5	16.4	-19.88	
10	16.2	16.6	17.6	-3.64	19.04
11	33.6	26.9	19.5	23.88	52.00
12	43.4	32.7	20.6	39.41	71.04
1	43.4	32.7	20.6	39.41	71.04
2	33.6	26.9	19.5	23.88	52.00
3	16.2	16.6	17.6	-3.64	19.04
4	5.9	10.5	16.4	-19.88	0.00
5	5.9	10.5	16.4	-19.88	
6	5.9	10.5	16.4	-19.88	

Summary:

Average Heat Pipe Input = 65.33 W
Mean Evaporator + Condenser Length = 1.8 m
Heat Pipe QL Product = 117.6 W.m
Heat Pipe Fluid Mean Temperature = 16.6 °C

thermal conditions for the ultimate solar absorption with zero power generation.

Using the heat pipe fluid temperatures determined in the ring analyses, the distribution of temperatures across a solar panel are shown in Table 3 and 4, again for the two solar angles and two levels of solar solar absorptance. The benefits of good thermal design are shown in the modest 30°C temperatures of even the most distant solar cells for the full power output case. The cell temperatures rise to near 45°C for the zero power output case, but then solar cell output efficiency is not an issue in this situation and the cells can tolerate the 45°C temperatures.

**Table 3. AMSAT Phase IV Spacecraft
Solar Panel Temperatures.
Degree Celsius**

Panel Modeled In Cross Section for 41mm Length of Heat Pipe

Solar Cell Absorptance = 0.719
Azimuth Angle THETA = 0 Deg.
T space = 4K
Absorbed Solar Heat, Qs = 4.350 W

Solar Cell Nodes

T11	T12	T13	T14	T15
29.0	25.4	27.2	25.7	29.4

Solar Panel Laminate Outer Skin Nodes

T20	T21	T22	T23	T24	T25	T26
19.1	19.0	15.1	17.1	15.5	19.5	19.5

Solar Panel Laminate Inner Skin Nodes

T31	T32	T33	T34	T35
18.1	15.0	16.0	15.4	18.5

Heat Pipe Nodes

T42	T43	T44
8.3	7.3	8.4

Heat Pipe Fluid Temperature, Thp = 5.1 °C
Heat Input To Fluid, Qhp = 2.158 Watts

Azimuth Angle THETA = 15. Degrees
Tspace = 4 K
Absorbed Solar Heat, Qs = 4.202 Watts

Solar Cell Nodes

T11	T12	T13	T14	T15
29.9	26.5	28.2	26.7	30.2

Solar Panel Laminate Outer Skin Nodes

T20	T21	T22	T23	T24	T25	T26
20.6	20.6	16.9	18.7	17.1	20.8	20.8

Solar Panel Laminate Inner Skin Nodes

T31	T32	T33	T34	T35
19.7	16.8	17.6	17.0	19.9

Heat Pipe Nodes

T42	T43	T44
10.1	9.0	10.1

Heat Pipe Fluid Temperature, Thp = 6.8 °C
Heat Input To Fluid, Qhp = 2.165 Watts

Conclusions

Designing a satellite is a multi-disciplined process of evaluating many different requirements, needs and materials. By no means is the Phase IV spacecraft design complete; a great deal of work remains to be done. The interactive processes that cross many technological boundaries will continue, as they must, without any one such technology dominating another without proper consideration of the success of the mission as a whole. It is hoped that this discussion has provided the reader some insight into the efforts needed to achieve a satisfactory satellite.

**Table 4. AMSAT Phase IV Spacecraft
Solar Panel Temperatures
Degree Celsius**

Panel Modeled In Cross Section for 41mm Length of Heat Pipe

Solar Cell Absorptance = 0.844
Azimuth Angle THETA = 0 Deg.
Tspace = 4 K
Absorbed Solar Heat, Qs = 5.065 W

Solar Cell Nodes

T11	T12	T13	T14	T15
43.4	39.5	41.7	40.0	44.1

Solar Panel Laminate Outer Skin Nodes

T20	T21	T22	T23	T24	T25	T26
31.8	31.9	27.6	30.0	28.2	32.6	32.5

Solar Panel Laminate Inner Skin Nodes

T31	T32	T33	T34	T35
30.8	27.5	28.7	28.1	31.5

Heat Pipe Nodes

T42	T43	T44
19.4	18.1	19.4

Heat Pipe Fluid Temperature, Thp = 15.4 °C
Heat Input To Fluid, Qhp = 2.648 Watts

Azimuth Angle THETA = 15. Degrees
Tspace = 4 K
Absorbed Solar Heat, Qs = 4.893 Watts

Solar Cell Nodes

T11	T12	T13	T14	T15
44.4	40.7	42.5	40.8	44.6

Solar Panel Laminate Outer Skin Nodes

T20	T21	T22	T23	T24	T25	T26
33.5	33.7	29.6	31.6	29.7	33.9	33.7

Solar Panel Laminate Inner Skin Nodes

T31	T32	T33	T34	T35
32.6	29.4	30.3	29.6	32.8

Heat Pipe Nodes

T42	T43	T44
21.6	20.4	21.6

Heat Pipe Fluid Temperature, Thp = 17.8 °C
Heat Input To Fluid, Qhp = 2.508 Watts

References

- (1) Jan King, W3GEY, Vern Riportella, WA2LQQ, and Ralph Wallio, W0RPK, "OSCAR at 25: The Amateur Space Program Comes of Age," *QST* December 1986, pp 15-18; and "OSCAR at 25: Beginning of a New Era," *QST* January 1987, pp 41-45.
- (2) Jan A. King, "Phase Four Technical Study Plan," AMSAT limited distribution document, 1 September 1986.

- (3) "Military Handbook Reliability/Design Thermal Applications," *MIL-HDBK-251*, 19 January 1978. Chapter 12, p 405.
- (4) "Systems Improved Numerical Differencing Analyzer" (SINDA) a mainframe thermal network analyzer program tracing its roots to the late 1960s and early 1970s in work done for NASA by TRW Systems Group.

Keplerian Elements for Manned and Miscellaneous Missions

Satellite: mir
Catalog number: 16609
Epoch time:
Element set:
Inclination:
RA of node:
Eccentricity:
Arg of perigee:
Mean anomaly:
Mean motion:
Decay rate:
Epoch rev:

88032.88717099
47
51.6309 deg
50.3592 deg
0.0016787
46.0859 deg
314.1993 deg
15.75088096 rev/day
1.9623e-04 rev/day²
11232

Satellite: ajisai
Catalog number: 16908
Epoch time:
Element set:
Inclination:
RA of node:
Eccentricity:
Arg of perigee:
Mean anomaly:
Mean motion:
Decay rate:
Epoch rev:

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50.0146 deg
320.1689 deg
0.0011300
277.2304 deg
82.7245 deg
12.44369614 rev/day
-2.5e-07 rev/day²
5561

Satellite: salyut-7
Catalog number: 13138
Epoch time:
Element set:
Inclination:
RA of node:
Eccentricity:
Arg of perigee:
Mean anomaly:
Mean motion:
Decay rate:
Epoch rev:

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51.6102 deg
227.6093 deg
0.0001116
50.6998 deg
309.4053 deg
15.31931674 rev/day
2.884e-05 rev/day²
33118

The Radio Links to Phase III-D An Initial System Concept

by Dr. Karl Meinzer, DJ4ZC

(Reprinted from *AMSAT-DL Journal*, Vol. 14, No. 1)

(Translated by Don Moe, DJOHC/KE6MN)

ABSTRACT

P-III-D is the code designation for a subsequent generation of the present P-III satellites as proposed by AMSAT-DL. For the radio amateur, the most important characteristic of these new satellites will be the significantly improved signal strengths. The relatively involved antenna installations as currently needed for OSCAR-10 operation will then be unnecessary. In fact, the radio links can be improved to such an extent that mobile or even hand-held operation will be possible.

1.0 Introduction

The P-III-D design specifies an elliptical orbit so that world-wide activity is possible with just one satellite. In this case the technology also remains fundamentally comprehensible, especially since the proven orientation control system in OSCAR-10 and the same antenna geometry can be duplicated. The parameters for the elliptical orbit would be set more precisely, however, so that of the two daily eight hour access intervals at our latitude, at least one interval would conceivably occur during local evening hours.

Recently, in the United States the "Phase IV Project," a system of several geostationary satellites, has been under discussion. Since a continuous access of 24 hours daily has the highest priority there, they are willing to sacrifice the possibility of world-wide communications and to accept reductions in the achievable signal strengths. From the European point of view, it is more sensible to stay with satellites using the Phase-III orbit and gradually supplement them with geostationary satellites. In the long run, this results in a communications system which would permit world-wide contacts at any time.

A frequently mentioned advantage of the geostationary orbit is that antenna tracking is no longer necessary and relatively large antenna gains can be achieved economically. In amateur practice this argument is only partially valid, since generally the antennas are also to be used for other types of operation and thus rotors are indeed required. The actual problem is that many amateurs are restricted from building large antennas, with or without rotors. P-III-D offers a solution from the other side: when the radio links are so good that even very small low gain antennas are adequate, tracking is no longer necessary. AMSAT-DL hopes that this philosophy will attract a larger number of people to satellite communications.

It is doubtless not immediately obvious why better signal strengths are possible in the P-III orbit than in a geostationary orbit. This point will be discussed more exactly in section 5, but for the moment suffice it to say that, in the case of a geostationary orbit, the main lobe of the antennas has to lie diagonal to the spin axis of the satellite. As a consequence, either a mechanically counter-rotating antenna or a three-axis stabilized satellite is required. In regard to our current launch opportunities, the bulkiness of the first solution reduces the gain by more than 8 to 10 dBi. The second solution involves such a complicated mounting and deployment mechanism for the solar generator, that we dread to think of the development and risks.

The P-III satellites are relatively simple and well adapted to our launch opportunities; the main radiation lobe is along the direction of the spin axis, and antenna gains of up to 15 dBi are readily achieved. This argument could possibly change in the future however, if the electrical power were increased over that planned for P-III-D. In this case, three axis stabilized satellites are cheaper because the cost of the solar generator could be reduced by half.

2.0 Target Parameters of P-III-D

The improvement of the radio characteristics of P-III-D compared to OSCAR-10 is achieved mainly through higher antenna gain at the satellite and higher transponder power. Both aspects presuppose a bigger satellite.

The large number of users (100 simultaneous channels) requires a bandwidth that is only achievable on Mode-L. The nominal power generation was designed so that activity restrictions would only rarely be required. Current planning centers around a satellite with the following data:

Diameter	3 m
Height without antennas	1 m
Mass at launch	400 kg
Available electrical power (nominal)	175 W
Transponder power consumption (nominal) ...	150 W
Average RF output power of transponder	50 W
PEP output power of transponder	200 W
Antenna gain (70cm & 24cm each)	15 dBi
Bandwidth of transponder	500 kHz
System noise temperature of 24cm RX	300 K

3.0 The Earth-Satellite Radio Link

In earlier satellites the radio link to the satellite had virtually no influence on the achievable system performance. In the case of Mode L this is no longer correct, and under certain circumstances this link can be more limiting than the link satellite-Earth.

Is it therefore even possible to operate through P-III-D using simple stations? The path loss at 1269 MHz and 37,500 Km distance amounts of 186 dB. For the 300 K system noise temperature in the satellite, the noise power in the SSB bandwidth (2.4 KHz) correspond to -170 dBW. To overcome the path loss with a 20 dB (PEP) noise margin, 36 dBW (PEP) minus antenna gains at both ends is needed. (See Figure 1.) When an antenna gain of 15 dBi is assumed at the satellite end, the transmitter power on the ground must be 21 dBW (PEP), or 126 W PEP into an isotropic antenna. It is in fact enough if for example a mobile station radiates in the direction of the sky with a minimal elevation angle of 20 or 30 degrees. An antenna gain of approximately 5 dBi results and thus a transmitter power of 40 W PEP suffices, which is relatively easily generated nowadays using transistors.

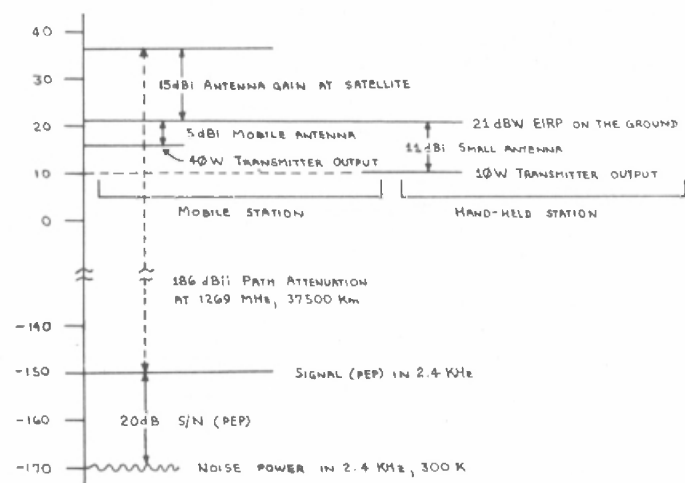


Figure 1: Earth-to-Satellite Path Loss.

Alternatively, a small helix antenna for a "hand-held" could easily provide 11 dBi, thus requiring a transmitter power of only 10 W PEP. From these numbers it can be seen that very little effort is needed to achieve a 20 dB noise margin on the upward link to the satellite. When it is considered that P-III-D is planned for the last decade of this century, it is entirely probable that manufacturers will offer appropriate equipment and antennas costing no more than the present-day 2m or 70cm SSB equipment.

On the uplink the problem frequently exists that too much transmitter power is being used, causing the transponder to reduce its gain which in turn forces all stations to increase their power beyond what would otherwise be needed. We can assume that incompetence, indifference, and poor receiving installations are the primary reasons that the power recommendations of AMSAT are not observed. We have, therefore, come

to the opinion that it is imperative to add certain technical features to the transponder in order to foster better understanding of satellite operation. As a reminder, the transponder in OSCAR-10 is almost always reduced in gain by approximately 15 dB; even on the QRP days, the limiter voltage is scarcely any less. In other words, if all stations would reduce their power by a factor of 30, their strength would not change one whit. Weaker stations would then also have a chance to use the satellite. In our experience it appears that appeals to self-discipline are quite futile.

During the time of OSCAR-7 a concept was already being discussed at AMSAT-DL which we named LEILA, for "LEistungLimit Anzeige" (Power Limit Indicator). This is a type of spectrum analyzer which searches out the strongest station in the pass band of the transponder and then, when the limiter voltage exceeds a prescribed value, inserts over this station a special CW marker signal. (See Figure 2.) Nowadays the implementation of this concept is relatively easy using the on-board computer. In fact it is possible to go beyond merely marking an excessively strong station by actually attenuating it immediately with a tunable notch, so as to prevent degradation to other signals.

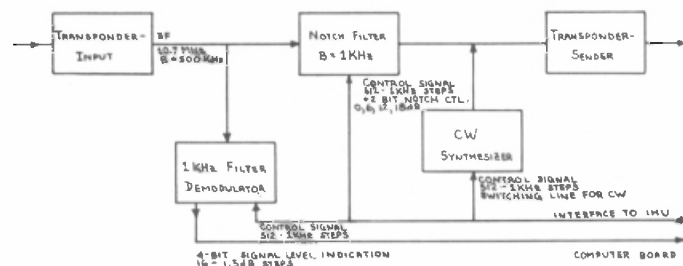


Figure 2: Power Limit Indicator.

In practice LEILA will operate as follows: When an inordinately strong signal appears in the transponder, it will be covered with a characteristic CW pulse. The station should then reduce power until the CW pulse disappears, resulting in the optimal power level. If a station should be much too strong, the notch filter quickly implements the power reduction. Due to the characteristics of the notch filter, the signal will sound "odd," in addition to having the marking pulse. In this case a reduction in power by a factor of at least 20 is warranted.

Since LEILA's spectrum analyzer is controlled by the on-board computer, several stations can be managed simultaneously. Suitable software can doubtless be created such that limiting in the transponder can be virtually eliminated. We have therefore definitely decided to include LEILA as a component of P-III-D. This measure, along with the high antenna gain, should assure that the satellite is truly accessible on a continual basis even for QRP stations. One other point that needs to be mentioned is that through computer control, special frequencies may be allocated a higher power level, thus providing preferential channels with exceptionally strong signals for emergency communications to any signifi-

cant extent. We hope that all users of the satellite will profit from LEILA. (The losers will be the manufacturers of superfluous amplifiers).

4.0 The Satellite-Earth Radio Link

While the noise margin on the upward link is determined by the effective radiated PEP power of the ground station, the satellite's average power per user is the appropriate measure on the downward link. When the average output power of the transponder is 50 Watts, this means that each of 100 active channels has effectively 0.5 Watt available.

For SSB the peak power (PEP) is approximately 10 times higher, i.e. the transponder provides nearly 5 W peak power per channel. Although this adds up to 500 W for the 100 channels and the transponder is capable of 200 W PEP maximum, this is no problem since not all channels require the peak power simultaneously. Even for as few as 10 channels, the fluctuations average out to the extent that the transponder can provide practically any peak power level to a single channel.

At 435 MHz and 37,500 KM distance, the path loss amounts to 177 dBii. With an antenna gain of 15 dBi at the satellite and 0.5 Watts of power per channel, the effective radiated power of the satellite is 12 dBWi, resulting in levels of -165 dBWi or -160dBW into a 5 dBi antenna on the ground. The noise power in the receiver is again assumed to be -170 dBW. This results in a 10 dB average noise margin, or 20 dB S/N PEP. The links to and from the satellite thus provide the same 20 dB noise margin for the minimal station setup assumed. (See Figure 3.) Even considering that both noise magnitudes accumulate and that small additional losses are unavoidable, the PEP noise margin should not fall under 15 dB; thus easily readable signals are available to the amateurs.

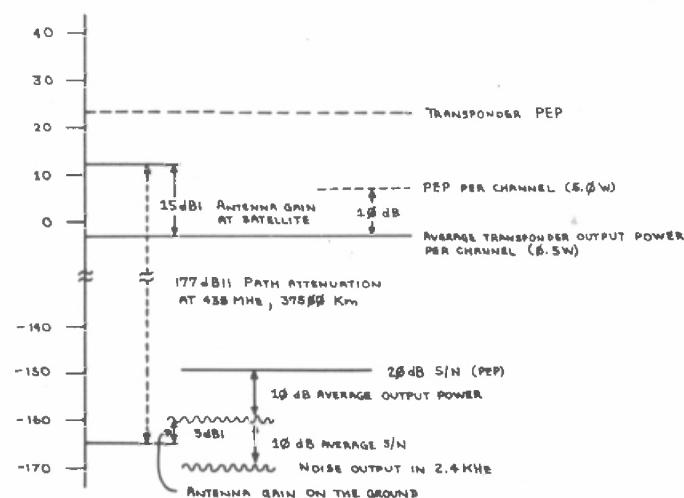


Figure 3: Satellite-to-Earth Path Loss.

In practice the two links will always differ by a few dB. For anything more than the minimal investment on the receiving side, the upward link will nearly always be the poorer one. The system therefore works at a level

where an even larger satellite could no longer boost the achievable noise margin.

5.0 Special Aspects of the P-III-D Satellite

5.1 The Orbit

Because of the low inclination angle of OSCAR-10, a completely false impression of the capabilities of an elliptical orbit has unfortunately developed in many minds. When the orbital plane is inclined approximately 60 degrees, both orbits are accessible at our latitude during a single day. In addition to the commonly known pass in a southerly direction until overhead, the second daily pass is heard in the northerly direction. In a manner of speaking, one looks past the north pole to the other side of the earth. Consequently there are only two brief interruptions of operation daily, while the satellite passes through perigee.

In P-III-D an electrically powered thruster will be incorporated which will permit alterations in the orbit even after operation commences. The orbital period can thereby be set to exactly 12 hours so that the orbit is exactly repeated day after day. Unfortunately the East-West drift of the ascending node remains unaffected, resulting in the geometry of visibility changing during the course of a year. Alternatively, there is also the possibility of positioning the longitude of the ascending node, which precludes the orbit from remaining synchronized with the clock. At this time it is still uncertain whether these two measures can be combined; there will surely be a few surprises encountered while tweaking the orbit.

5.1 The Antennas

The original P-III-D design implements antennas with approximately 10 dBi of gain. At apogee, the furthest distance away, these antennas are directed optimally at Earth; at other points along the orbit, the antenna squints past the Earth. As the distance to the Earth decreases, the field strength remains nearly constant over a large portion of the orbit. (See Figure 4.)

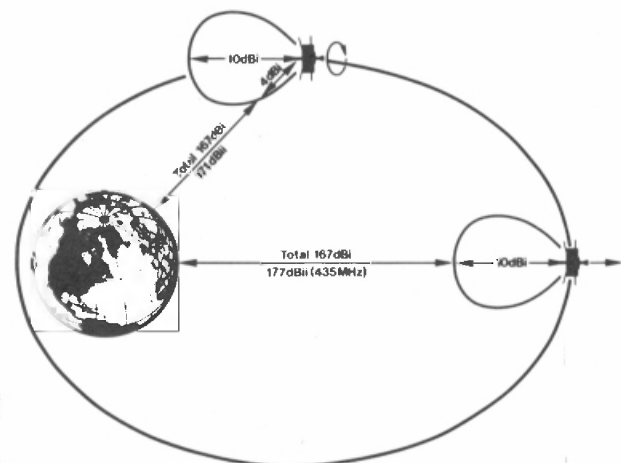


Figure 4: Phase III Satellite Antenna Pattern. (Example for 435 MHz)

Theoretically it is possible to increase the antenna gain to around 18 dBi; higher gains are not possible since the main lobe becomes too narrow and no longer encompasses the entire globe. As a consequence of such a high gain, signals rapidly fall off in strength away from apogee, and only a fraction of the orbit is useful. When an antenna gain of 15 dBi is selected, the conditions improve, but a large unusable portion of the orbit remains. Since P-III-D is rather large, the opportunity arises to synthesize the antenna patterns such that the radiation pattern can be changed during orbit providing optimal gain at any given point. (See Figure 5.)

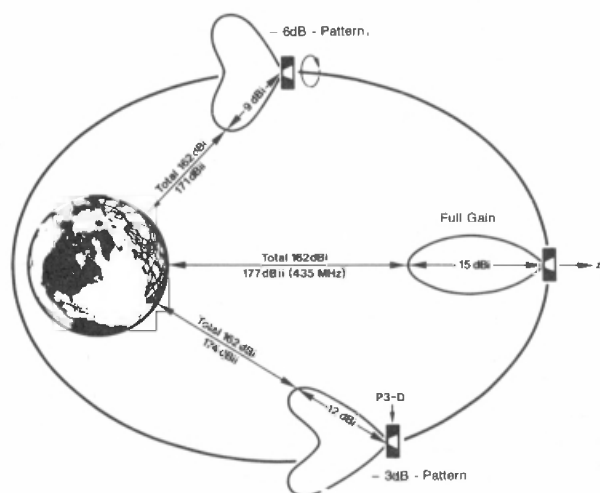


Figure 5: Phase III-D Satellite Switchable Antenna Patterns. (Example for 435 MHz)

It should be noted that all patterns must always be rotationally symmetrical about the Z axis in order to avoid spin modulation. The switchable patterns make possible an antenna gain 5 to 10 dB higher than the original P-III antennas.

5.3 The Power Supply of P-III-D

Under ideal circumstances the solar generator of P-III-D can supply approximately 300 W from its 6 square meters of cell area. When a 20% reduction due to aging from radiation and a sunlight angle of 30 degrees are considered, the available power is reduced to a bit under 200 W on the 28 V bus. Of that, 25 Watts are consumed by the satellite itself for propulsion, position control, and on-board systems so that at least 150 W remain for the transponder. Using this conservative calculation, no limitations on communications should be necessary during the majority of the satellite's life expectancy.

To permit uninterrupted operation during eclipses, the battery should have enough capacity that it can supply the power of the solar generator for at least three hours. In OSCAR-10 this condition is partially met. In the case of P-III-D, this means that 750 Watt-hours must be stored. At this time it is uncertain whether it is really prudent to provide such a large battery weighing nearly

40 Kg. It is probably more reasonable to select the battery based on the needs of the propulsion system, nearly 400 Watt-hours, and to accept interruptions to communications or reductions in power during the eclipses. On the other hand, there is no longer the capability of storing enough energy during perigee to enable full operation during the remainder of the orbit despite unfavorable sun angles.

6.0 Conclusion

As described, stations having only limited antenna gain will be able to use the satellite fully based on the current design parameters for P-III-D. This will establish satellite communications on a dramatically broader base, since the antenna expense for the first generation of P-III satellites has been regarded as being the main deterrent to this mode of operation. In fact for the first time, stations interested in world-wide DX communications via P-III-D will be significantly cheaper than their functional counterparts on short wave. The modest requirements for antennas will make the satellite interesting to those stations that cannot erect outside antennas, and the transmissions on 23 cm can hardly cause any TVI at the relatively low power levels used.

The link improvements of at least 10 dB compared to OSCAR-10 are achieved through the size of the satellite and its high transmit power of 200 W PEP. Additionally the new antenna concept for switchable radiation patterns and the electronic notch to prevent misuse by strong stations contribute considerably to this design.

Some Thoughts on RUDAK Traffic Control

by Hanspeter Kuhlen, DK1YQ
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(Reprinted from *AMSAT-DL Journal*, Vol. 14, No. 2)
(Translated by Don Moe, DJ0HC/KE6MN)

The technical characteristics and possibilities of the digital transponder aboard Phase 3C have already been described in various articles, including those appearing in this journal. This article presents some thoughts about the operation and effective use of the transponder.

The unavoidable and inefficient Aloha access to the transponder along with the low data rate (throughput) of 400 bps requires some type of traffic management if optimal operation is to be achieved. The data throughput is nothing more than the sum of all packets which actually pass through the transponder. As illustrated in Figure 1, starting with just a few packets, the throughput initially expands with increasing packet density. This continues until the retry rate due to collisions becomes so large that new packets scarcely have a chance to be retransmitted. If the packet density increases beyond this point, the channel becomes so completely obstructed that ultimately no packets come through. The goal of traffic management is therefore to maintain the optimal activity level.

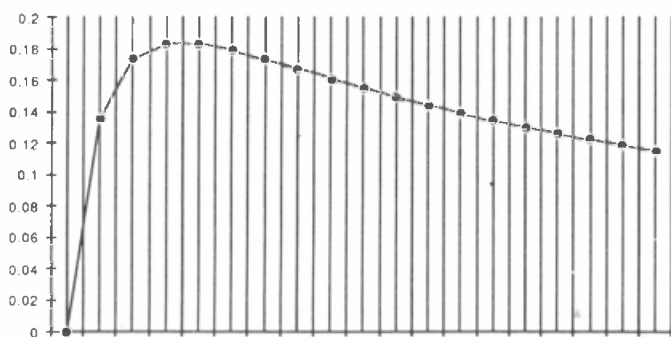


Figure 1. Throughput for ALOHA Systems.

Naturally, this kind of management cannot be attained in amateur radio through the use of elaborate control stations as is in the case for the commercial satellite-based data transmission systems. Such stations allocate exact time slots to each participating station corresponding to the traffic volume according to a predetermined plan. If merely the more important functions of such a control station were to be emulated, an elaborate protocol and an even more complex synchronization technique would be required for additional stations joining the net. A significant complication is caused by the 134 msec propagation delay to the satellite at apogee (fur-

thest point on an elliptical orbit of approximately 39,000 KM from the earth). Such precise scheduling, involving strict observance of exact transmission intervals at which the next packet could be sent, are totally impractical using amateur resources. Aside from that, the stringent regimentation would be neither desirable nor enforceable. Thus the only appropriate technique is one in which the scheduling is performed by the satellite autonomously.

For this reason a modified form of the Aloha method of traffic control will be evaluated during RUDAK operation. In this method, access is dynamically restricted depending on traffic volume. Before describing the type of traffic control using a so-called "list," a discussion of RUDAK's transmission capacity is necessary.

The limitation on the total throughput is caused by the low data rate of 400 bps on the downlink. This data rate is not limited by RUDAK itself, since RUDAK could also process 9600 bps or even higher without change. The value of 400 bps was selected during RUDAK system design from calculations based on the radio link and the efficiency of a typical amateur radio station and its achievable system characteristics (transmission power vs. receive sensitivity). Those who have observed operation via OSCAR-10 will confirm that the paramount problem for most amateur radio stations is optimizing the receiver sensitivity, and to a much lesser degree, increasing transmitter power.

The declared goal of the RUDAK experiment is to make available to individual users an experimental channel for direct contacts and tests. Higher capacity channels supporting 9600 bps or even 64 kbps naturally sound much more fascinating than the 400 or 1200 bps that RUDAK makes possible. In this connection it shouldn't be overlooked, however, that even commercial manufacturers are currently expending a great deal of effort on equipment for these data rates and are having their problems. In the case of amateur radio, it's not the business prospects, but rather the technical aspects which (still) stand in the foreground, i.e. to derive the maximum knowledge and potential using the simplest means or to prove the fundamental feasibility of a concept. It is therefore irrelevant whether RUDAK can be sensibly used for exchanging megabytes of data between two computer centers or not.

Under these aspects operation via RUDAK at the transmission rate of 400 bps is still quite attractive for two stations. However, in the case of only five simultaneous connections involving ten stations with statistically evenly distributed transmissions, the data rate drops to an average of 40 bps (or 120 bps respectively). This speed corresponds approximately to the transmission speed of RTTY, though it is error free in comparison.

As already mentioned, since the level is reached when the channel is already operating at essentially full capacity, a further increase in the number of stations (or packets per station) would disproportionally decrease the throughput. Where this optimal level lies depends upon several factors (protocol parameters), which are to be determined and confirmed by the RUDAK experiment. An important parameter in this connection, is for example, the maximum packet length. This will be initially set to 128 bytes, but it and the other related parameters can be adjusted in orbit by the command stations as experience is gained.

Being able to try out various parameters is certainly an eagerly awaited part of the RUDAK experiment. We hope to be able to report soon on our efforts at optimizing these parameters. The suggested settings will be broadcast by RUDAK in its beacon packets (UI frames), so that everyone can adjust the TNC parameters accordingly. Everyone is invited to actively participate in the experiments. Here is a good opportunity to learn the basics and techniques of remote data processing.

Now back to traffic management. The procedure should react dynamically to changing activity levels. In order to achieve this goal, the concept of automatically maintaining a "list" has been suggested. Each station (call sign) which wishes to send via RUDAK must be on the list; otherwise its transmissions will be ignored. Expressed another way, he who isn't on the list doesn't need to send any longer since the frequency is already at its optimum capacity and an additional station would merely make the channel unusable for everyone. This way the number of accessing stations will be automatically limited to the quasi-optimal number.

The equivalent to this condition in the frequency domain (frequency multiplexed access) is presumably more familiar. An amateur band which is essentially full with stations cannot be used by any additional stations, since no vacant frequencies are available. Correspondingly for time multiplex access, there are no more vacant time slots. The exact number of list places, presently estimated at 10, will be adjusted according to experience gained in actual operation.

RUDAK distinguishes between only two conditions:

1. Normal Mode = Digipeat and Robot during light loading.
2. List Mode = Digipeat limited and no Robot during heavy loading.

In other words, during times of light activity RUDAK acts as a normal digipeater, preparing and retransmitting on the downlink everything received on the input frequency (1269.675 MHz) in the AX.25 protocol. The

digipeat mode therefore doesn't require any further comment since it doesn't differ significantly from the familiar terrestrial operation.

As soon as the activity level requires the list mode, no further UI packets will be sent. In their place, a list will appear at intervals still to be determined. The entire remaining time is therefore available for use by the admitted stations. "Admitted" in this context are the ten or so last call signs which were already in QSO at the moment of switch-over. New call signs can only be added to the list when free positions become available.

The list is hence a type of storage routine which analyzes all arriving packets in a given time interval and determines the current status of call signs and packet quantity. A new station (call sign) is only admitted when it is added to the list by the satellite following a transmission attempt. This again is only possible when a vacant place is available.

To illustrate the dry theory, here are a few examples.

Normal Mode:

The satellite is sending packets. An OM who is copying this activity will likely see something like this:
(The example refers to a TNC1 with the WA8DED multiconnect software. For other TNC versions, the display will differ accordingly.)

fm RUDAK to QST ctl UI pid 50
This is general information from AMSAT to all Phase 3C users, etc...

fm W3GEY to DJ4ZC ctl I10 pid fO
hi Karl, how are you?

fm DJ5KQ to G8DQX ctl I12 pid FO
Robin, thank you for the QSO, 73 de Werner

Operation is essentially the same as it currently occurs terrestrially. A peculiarity, though, is that RUDAK doesn't appear with its call sign as digipeater.

Robot Operation:

This type of operation is only activated in the Normal Mode. It is designed to provide as quickly as possible a thoroughly adequate answer to the question whether the selected station parameters are correct, thus permitting proper operation via RUDAK.

For example, if I send the following packet:

* CONNECT RUDAK

and everything is correctly set, immediately (270 msec) afterward the message:

*** RUDAK Busy
DISCONNECTED from RUDAK

appears. Thus I know beyond any doubt that in addition to my RF path being in order, the AX.25 protocol and the corresponding parameters are correct and that I can carry on a QSO via RUDAK. Besides this check function for verifying one's own radio path, there will be an additional service provided.

If I send in place of the RUDAK call sign:

* CONNECT RUDAK-n (for n = 1 to 5)

RUDAK answers me "personally" with predetermined standard information. For example:

* CONNECT RUDAK-1

fm RUDAK to DK1YQ ctl I01 pid f0
Current Kepler orbital parameters...

Or

* CONNECT RUDAK-2

fm RUDAK to DK1YQ ctl I01 pid f0
Orbital data:

-1234 Orbit: 235 MA:80/10 14:35UT
DOP:-2300Hz Attitude:Long/Lat/Spinrate

73 de RUDAK

N.B.: MA80/10 indicates that the mean anomaly is presently 80 (of 255) and RUDAK operation is possible for another 10 MA counts (approx. 25 minutes) because Mode-L will be switched off, for example. RUDAK is unfortunately only operational in Mode-L. Following the satellite's time in UTC is the actual Doppler frequency shift at this position, ie. the amount that I must correct my transmit frequency to compensate for the effects of Doppler shift. Finally appears the present attitude of the satellite and thus of the antennas in longitude, latitude, and spinrate, referenced to the orbital plane. Additional possibilities:

* CONNECT RUDAK-3

sends some selected telemetry values.

* CONNECT RUDAK-4

provides the current transponder operating schedule for Phase 3C.

* CONNECT RUDAK-5

supplies brief information from AMSAT regarding eclipse times, etc.

These are all immediately followed by

*** DISCONNECT fm RUDAK

List Mode:

RUDAK List: W3GEY + DJ4ZC + DJ5KQ +
G8DQX + DL2MDL + JR1SWB + DF8CA +
WBORLY + .. + ..

fm W3GEY to DJ4ZC ctl I10 pid F0
fm DJ5KQ to G8DQX ctl I01 pid F0
fm DL2MDL to JR1SWB DISC

To get onto the list is a "simple" problem. It is much more difficult to define the criteria by which a place in the list can be freed.

A time-out procedure seems to be the simplest solution to this problem wherein the connection is terminated for each station which hasn't sent a packet during a specified time interval. It would clearly be disadvantageous to encourage superfluous retries or empty packets merely to remain on the list.

Another feasible alternative is the evaluation of the corresponding AX.25 control field where the disconnect request is apparent in the DISC bit pattern. In this instance, however, both stations must continue to have access to the satellite until the end of the connection since their call signs would occupy valuable spots on the list for an unnecessarily long time. Finally, it would be conceivable to limit the time on the list to perhaps 10 to 15 minutes.

In any case, the operation should automatically switch back to Normal Mode from List Mode as soon as the activity level permits.

At this time these questions are being intensively investigated and tried as part of the RUDAK field test from the water tower in Ismaning (DL0ISM) near Munich.

The following literature contains further information to the theme Packet Radio and the RUDAK experiment:

- (1) T.L. Fox, WB4JFI, "AX.25 Amateur Packet Radio Link-Layer Protocol," Version 2.0, October, 1984, ARRL, Newington, CT USA.
- (2) H. Kuhlen, DK1YQ, "The RUDAK Experiment of AMSAT-DL," *AMSAT-DL Journal* No.1 and 2, Jan/Feb and Mar/Apr 1986.
- (3) H. Kuhlen, KK1YQ, "Packet-Radio—A New Mode of Operation," Digital Transmission Methods in Amateur Radio, 1st Munich Packet Radio Conference, Munich, 30 March, 1985.
- (4) H. Kuhlen, DK1YQ, "Digital Packet Radio Transmission," Report from the ARRL ad hoc group for digital PR communication from October, 1984, *CQ-DL* 1/85.
- (5) Dr. K. Meinzer, DJ4ZC, H. Kuhlen, DK1YQ, "Report on the Formal Definition Meeting for the Packet Radio Experiment to be included in AMSAT-P-III-C satellite," 4th ARRL Amateur Radio Computer Networking Conference, San Francisco, CA, 30 March, 1985.
- (6) R.E. Raikes, WA8DED, "AX.25 Version 2—Multi Channel TNC Firmware," translated and reworked in 1985 by H. Giese, DF2AU, P. Gulzow, DB2OS.
- (7) H. Kuhlen, DK1YQ, "Protocol: RUDAK Design Review Meeting," Marburg, 19-21 July, 1985, *AMSAT-DL Journal*, Nr 31/32, 1985.
- (8) H. Kuhlen, DK1YQ, "Report on the Planned AMSAT Phase IIIC Satellite," 1st UHF Congress of the OVSV Ried, Austria, July, 1985.
- (9) H. Kuhlen, DK1YQ, "Introduction to the RUDAK Experiment," *73 Magazine*, August, 1986.
- (10) "RUDAK Users Manual," presently in preparation.

PSK Interface for the TNC1

by Peter Gulzow, DB20S

(Reprinted from *AMSAT-DL Journal*, Vol. 13, No. 6)

(Translated by Don Moe, DJ0HC/KE6MN)

ABSTRACT

The following article describes an interface to the AFREG PSK demodulator needed for operation through RUDAK, the packet radio experiment aboard AMSAT-DL's Phase-3C satellite to be launched this coming year.

1.0 Introduction

Over the past several weeks, there have been numerous inquiries regarding the articles on the AFREG 400 bit/s PSK demodulator. Packet radio operation via the RUDAK digipeater requires this demodulator as a prerequisite for receiving the 70 cm PSK transmissions. To send to the digipeater, it is necessary to generate a 2400 bit/s PSK signal on 23 cm. A corresponding circuit was published in the previous issue of the *AMSAT-DL Journal* (Vol. 13, No. 3, Sept/Oct 1986). A small additional interface, described in this article, makes the signals palatable to the TNC.

2.0 The Terminal Node Controller

The term "Terminal Node Controller" refers to a converter for transforming packet radio transmissions (AX.25 Level 2 Version 2 protocol) into a conventional asynchronous ASCII data stream which any terminal or computer with a serial RS-232 interface can receive. Because of the full duplex operation with unequal baud rates and PSK modems, only the TNC1 and TNC2 packet controllers from TAPR are suitable for RUDAK operation. Several manufacturers market compatible clones of the TNC1 and TNC2. Various other packet radio controllers and software solutions are not suitable since they do not support full duplex operation with unequal baud rates or offer an appropriate connection for an external modem. Incidentally, the ground station equipment for PSK telemetry decoding and antenna tracking based on the Atari 800XL is not capable of operating packet radio via RUDAK. In the case of any doubts about the applicability of a certain TNC for RUDAK operation, the members of the RUDAK group in Munich or Hannover may be able to provide an answer.

3.0 The PSK Interface

The circuit shown for the PSK interface was especially

designed for connection to TAPR's TNC1 or compatible. A similar circuit is needed for the TNC2, although the so-called clock synchronizer would be omitted. In addition, a modification to the TNC1 software to trap an error in the WD-1933/35 HDLC controller is absolutely required.

In the transmit section of the PSK interface, the bit clock is derived from the internal TNC clock by a 1/32-divider and is fed to the HDLC controller's transmit clock input. Following differential encoding, the transmit data stream is combined with the bit clock and sent to the PSK modulator at 2400 bit/s.

A second output is provided to permit 400 bit/s operation over the analog transponder using a conventional SSB transmitter. In this case, the transmit data stream is also mixed with an auxiliary carrier. The 1/8-divider output provides a frequency of 1600 Hz, which is combined in an XOR gate with the 400 baud transmit data. This auxiliary carrier, now modulated with PSK, is fed to the audio input of the SSB transmitter via a low-pass filter. Due to the required bandwidth, this technique is not suitable for the 2400 bit/s uplink to RUDAK.

Switch S1 selects the type of differential encoding. In AMSAT format, a logical "0" is transmitted when two successive bits have the same level, either "00" or "11." A logical "1" is transmitted when a transition occurs, "01" or "10." The AMSAT format is used on the RUDAK 400 bit/s downlink, the RUDAK 2400 bit/s uplink and the PSK telemetry.

In the case of the NRZI format, the definitions are exactly reversed. A logical "0" corresponds to a transition in the data stream, and a logical "1" indicates that there is no transition in the successive data bits. The NRZI standard is used for the RUDAK and JAS-1 1200 bit/s downlinks and for packet radio in general.

The receive section of the interface removes the differential encoding from the data stream coming from the AFREG PSK demodulator and feeds the resulting signal to the HDLC controller in the TNC. The receive clock from the demodulator is additionally resynchronized using the transmit clock before being sent to the HDLC controller, to compensate for a problem in the WD-1933/35. As in the transmit section, either AMSAT or NRZI format may be selected by S2.

NUSAT-I's "Layered" Protection Software Design

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ABSTRACT

This paper describes the approach used aboard NUSAT-I to provide protection against catastrophic algorithmic errors in uploaded command software. From that perspective, it also discusses the potential of a philosophy of "operation" rather than "use" for future AMSAT spacecraft.

1.0 Introduction

On April 29, 1985, NUSAT-I (Northern Utah Satellite) became the first satellite ejected from a Space Shuttle Get-Away Special Canister. The flight was Challenger's Mission 51-B with ejection taking place at an altitude of about 350 km. and an equatorial inclination of 57 degrees. The satellite was constructed by undergraduate students at Weber State College in Ogden, Utah under the direction of faculty and local engineers who volunteered their time.

With a launch fee of \$10,000 and a cash construction cost of less than \$20,000, NUSAT-I's design and manufacturing philosophy was prototypical of what are now being called "cheapsats." Despite this low price tag, however, cheapsats need not be trivially capable. NUSAT-I was equipped with an NSC-800 computer which was fully programmable by software uploaded from the ground station at Weber State College. The satellite was also equipped with the sensors necessary to perform its mission, which was to measure the amplitude pattern of Federal Aviation Administration radars. A coincident goal, strongly supported by FAA, was to provide a unique educational experience to students working on the project.

This paper will describe in detail the design of the on-board software with emphasis on those features that protected the satellite from execution of uploaded software with catastrophic algorithmic errors. AMSAT satellite designers may find this description of particular interest as they progress past the point where Amateur Radio spacecraft are provided for "use" and on to the point where, perhaps, they might be provided for "operation." Recent proposals of the EDSAT type suggest the possibility of one day having users (students?) on the ground program experiments aboard Amateur Radio satellites. This sort of activity would bring the excitement of working with spacecraft closer to the general public. Indeed, AMSAT satellites and their manipulation might serve as a new focus for generating interest in Amateur Radio at the high school and university level.

2.0 NUSAT-I's Software Design

In designing the software for NUSAT-I, two primary considerations drove our approach. These were operational versatility and spacecraft safety.

We knew almost from the beginning we were going to build a satellite that would be programmable post-launch from the ground. This was compelled by the need for versatility in performing the FAA's experiments. The sheer complexity of those experiments made it clear that a complete pre-launch design of algorithms was not feasible. Instead, we expected the development process to be evolutionary. Improved versions of programs would be defined by the results obtained from previous versions.

We were also sensitive to the other consideration, spacecraft safety. We knew that the crop of students working on the early designs were not going to be around for the entire mission. In fact, on the day when the first lines of code for the on-board operating system were assembled for the EPROM, the people who would eventually be writing software for upload were still in high school.

We were especially conscious of this, but it eventually became clear that the design would have had to maximize safety regardless of the experience level of the software team. Allowing uploaded software to accidentally end the mission was an intolerable possibility, regardless of who wrote it. Consequently, the design of the operating environment within NUSAT-I's computer was arranged to prevent commands being issued to the peripheral hardware which could drain and damage the batteries or, worse, place the system in a permanent loop from which it could not respond to ground commands.

With this defensive attitude in mind, a concept of "layering" emerged from our designers which provided as much fool-proofing as was thought possible without degrading the versatility requirement. The "layers" refer to event thresholds in the operating system that, when triggered, returned the computer's instruction pointer to the main idle loop.

3.0 The Processor

Before we examine the design in further detail, a brief description is in order concerning National Semiconductor's fine NSC800 system, which they donated for NUSAT-I. The NSC800 is a Z-80 programmed, 8-bit microcomputer designed for low-power applications. In the arrangement we used on NUSAT-I, there were 2K

bytes of ROM and 2K bytes of RAM on the CPU board. 48K of additional RAM was also available which could be powered up or down on three boards of 16K each. Figure 1 shows a detailed memory map of how our architecture was laid out.

0000h to 07FFh	— System ROM
0800h to EFFFh	— RAM for uploaded programs and data
F000h to F7FFh	— RAM for system use (clocks, flags, etc.)
F800h to FFFFh	— I/O port definition and addressing
001Dh	— Main loop start-up
0110h	— Exec vector jump (end of loop)
0120h	— POWER check subroutine
0187h	— MODCHG subroutine
0224h	— COMM subroutine

Figure 1: NUSAT-I's Address Map

Driving the microprocessor's NMI (non-maskable interrupt) pin was a pulse which occurred each 16 msec. It was from this that the system's internal software real time clock was computed. This clock was four bytes long providing many years of time-keeping. Much of the "layering" was in some way timing dependent. Figure 2 is a flowchart of the main loop of the onboard operating system and the non-maskable interrupt (NMI) processing routine which updated the software clock.

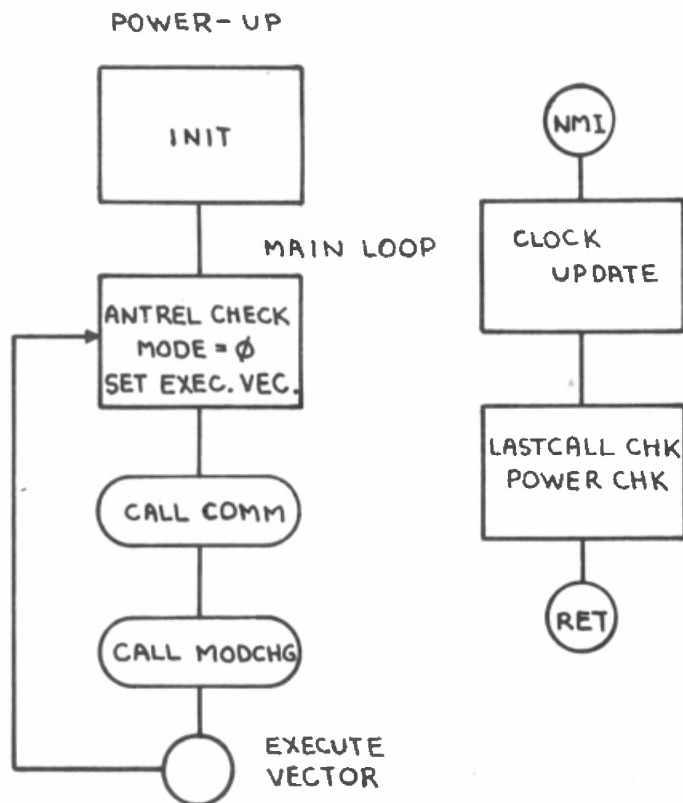


Figure 2: System Flowchart.

4.0 Operations

As can be seen in Figure 2, at ejection the code began execution with the initialization block which performed the sort of tasks one would expect. It set the clock to zero, switched off all equipment aboard (except the computer itself and the communications receiver, both of which were hard-wired on), began the antenna release clock, and initialized the I/O ports on the NSC810 peripheral I/O chip.

From there the main loop was entered. Within the main loop were calls to the antenna release code, the COMMunications subroutine and the MODCHG (MODeCHAnGe) subroutine. MODCHG was the generic power switching routine for the satellite's various groups of equipment. In the main idle loop, it commanded an unconditional sequence to Mode 0 (lowest possible power drain mode).

The fundamental approach to safety in the design was to resume execution of the main loop, providing the ground station with access to the satellite. As with any programmable spacecraft, access from the ground permits identification of problems and, hopefully, their correction. Each layer of protection therefore executed a return to the main loop whenever it was triggered. Notice that this action did not destroy the clock, nor did it reinitiate the antenna release sequence. The clock was set to zero only at start-up.

As a matter of interest, the last point is significant. In keeping with our low cost approach to construction, our antenna release mechanism was a loop of nylon fishing line around a short length of nichrome wire. A pulse of high amperage through the wire cut the nylon, melted the wire itself and allowed the antennas to spring out. Please note that nylon was not chosen randomly. Monatomic oxygen in the upper atmosphere causes nylon to deteriorate and so, in choosing that material, we provided ourselves with a measure of redundancy.

If the nichrome wire did not melt, we did not want current (about 20 amps) passing through it each time through the main loop. Hence, the release command was stringently constrained to occur only at certain times. The entire sequence was tested on the ground and pulse duration margins were carefully arranged. The release command was issued 17 seconds after ejection and once each 24 hours thereafter. The repetition was desired for confidence and at a daily frequency, it was not considered dangerous.

During the mission, we were able to get visual confirmation of antenna deployment from the astronauts. This observation told us much about the satellite's operational status and is a distinct advantage of the Shuttle-GAS launch alternative.

5.0 The Layers

Once in the main loop, the system idled until contact with the ground indicated a program was about to be loaded. After uploading, the new program executed and the layers became active.

The first of these involved a battery check clock kept inside the NMI routine. This clock ensured that a measurement of the battery voltage was taken at least once each ten minutes. We used Gates lead-acid cells aboard NUSAT-I recharged by state-of-the-art TRW solar cells. The solar power bus was configured to produce about 13 volts, which was sufficient to recharge our 10V batteries. The NMI power check did an unconditional jump to the main loop and Mode 0 at 9.6 volts, which equated to a rather serious discharge state. A student program which placed the satellite in a high power drain mode would therefore not result in catastrophe. The power check layer would trigger automatically within ten minutes of the 9.6V level and return to the main loop wherever in the orbit the satellite might be.

The next protection layer was in COMM, the subroutine that interfaced with the communications receiver. Whenever COMM was called, its first instructions placed the current time of day in a memory location labeled LASTCALL, meaning the time the routine was last called. Each time the NMI fired (each 16 msec.), it checked LASTCALL against the current time to see if ten minutes had elapsed. If so, a problem was declared and a sequence to the main loop and mode 0 was commanded. This would abort the uploaded problem program and power down the RAM in which it had been executing. However, if uploaded programs were calling COMM sufficiently often, the routine returned from interrupt and continued execution. Students were informed that they must call COMM every ten minutes no matter what their program was doing.

Some rather standard communications precautions were also taken to protect against erroneous software. Uploaded programs were checksummed by the on-board uploader routine before execution. A failed checksum yielded a failure acknowledge to ground and a jump to the main loop. The standard protections in the UART hardware (data format, parity, etc.) were also employed.

6.0 Testing

Generally, complex programs were not uploaded without prior testing of abbreviated versions. This refers specifically to those experiments intended to be performed throughout entire orbits while the satellite was out of contact with the ground station. These were tried over a period of two or three minutes while the satellite was accessible overhead so that programs could be halted without relying excessively on our layers.

Results of the shortened programs were checked before "turning them loose" for a full orbit. This proved especially true with the FAA radar experiments. The L-Band radar equipment aboard drew 3 amperes and had to be treated with great care.

I recently had an opportunity to discuss this concept of "a priori" testing with Dr. Martin Sweeting of the University of Surrey. The UoSAT's are fully programmable and the problem of testing has been accorded a great deal of attention in their program. Dr. Sweeting described their extensive simulation facilities to me and

pointed out that it was their policy in UoSAT operation to test thoroughly before upload. Then, after additional discussion, it became clear that this was in addition to the various protections they had built into their spacecraft.

This degree of thoroughness speaks well of the superb efforts we know take place at the University of Surrey, but it does pose a question concerning the theme of our approach on NUSAT-I and that we are urging in AMSAT satellites, i.e., operation (programming) by interested people on the ground who might not have access to simulators. The question is: can adequate protection be placed in the spacecraft operating system to make the owners willing to risk less-than-expert programming? We believe so, but the AMSAT community will have to subject it to debate to establish a consensus.

Let me close this article with our final layer of defense. After much thought we concluded that it was impossible to completely protect NUSAT-I with purely software techniques. Because we were depending on ten minute clocks in RAM to prevent loss of the satellite, we were not completely safe. The RAM locations containing those clocks could be corrupted by either uploaded software or by chip damage due to radiation. The solution we implemented addressed both potentialities.

It was simply a watch-dog timer designed to force a call to COMM every 16 minutes. Figure 3 shows its design. Once again, our goal was to ensure eventual ground access was guaranteed. A call to COMM would yield a certain bit pattern on the computer's address bus representing COMM's location in ROM. Simultaneously, an instruction fetch signal would occur from the processor. Each time this combination of events took place, the watch-dog timer would be reset. If the combination did not take place, i.e., COMM was not being called, the timer timed out. It would then pulse the reset pin on the NSC800 and the entire system would reinitialize to start-up. With this technique added to the other layers, we felt confident uploaded software could not destroy the satellite.

7.0 Requiem

NUSAT-I reentered the atmosphere and burned up in December of 1986 after having provided Weber State College students with an invaluable educational facility and the FAA with a new calibration device for almost two years. During the mission, there were numerous occasions when the protection layers just described were called upon to perform. On no occasion did they fail to do so.

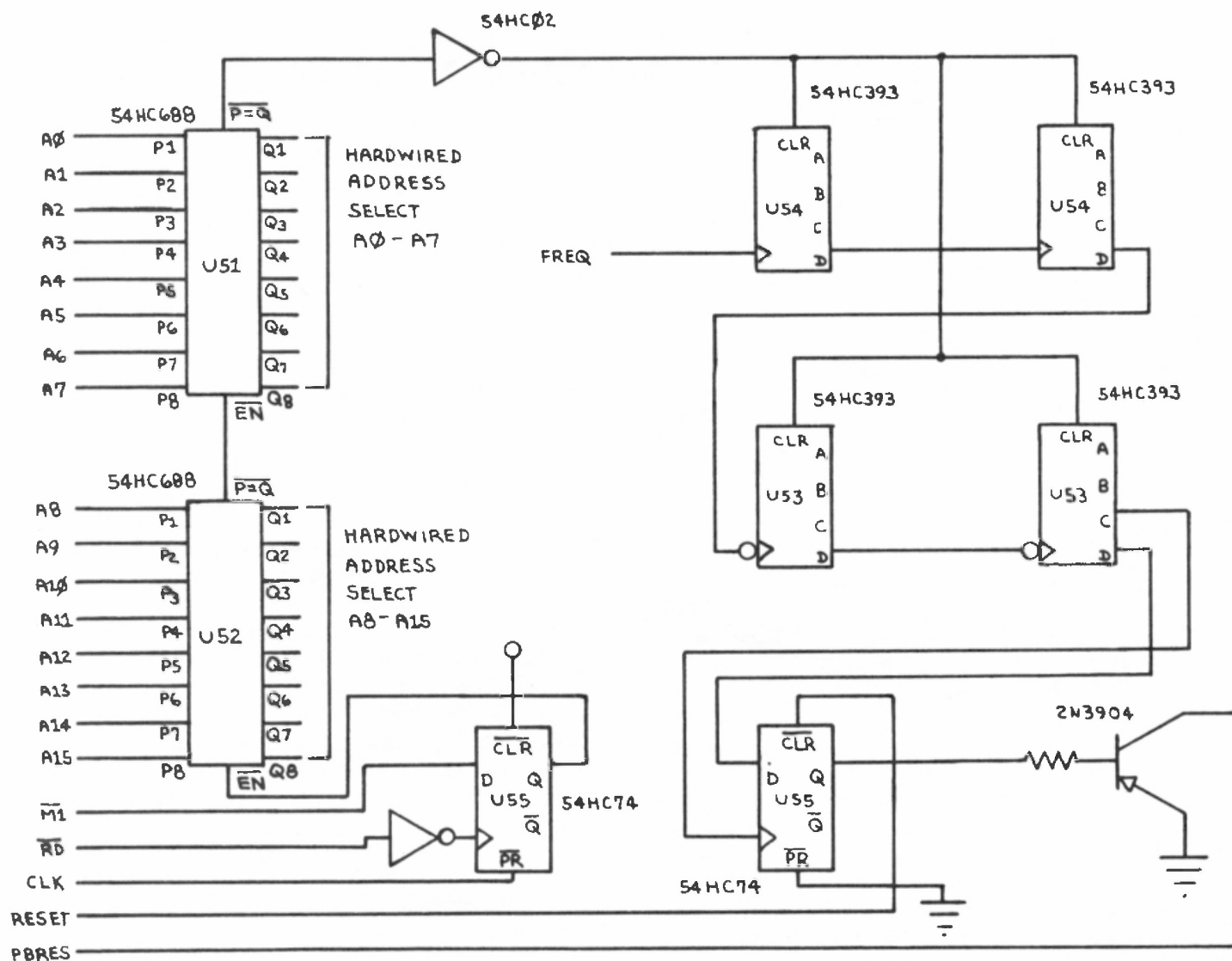


Figure 3: Reset Timer

A Method for Evaluating Antennas For a Low Earth Orbit Mission

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I. The Problem Addressed

Antennas for amateur satellites have traditionally been fabricated to fit into launch constraints and to deploy and operate properly in orbit. Mission profile has been considered in some antenna designs, such as that for the AMSAT Phase III satellites. Amateur satellite users are mostly experimenters who are willing to invest time and money to make up for non-optimum orbiting antennas with sophisticated ground station equipment. The service proposed for prototyping with the Packet Technology Satellite Experiment (PTSE) or "HouSat" is expected to require more from the satellite and less from each of a larger number of ground based users.

One way to get more from the satellite is to use some sort of optimized antenna. The goal of this paper is to provide a rationale for evaluating possible antennas for this service in terms of their radiation patterns. Selection of such an antenna system involves these considerations: electrical performance, launch constraints (envelope), and user-relative performance. This discussion primarily addresses user-relative performance desires. No particular antenna is recommended but several possibilities are discussed as a starting point for prototyping.

The ideal antenna would provide equal radiation coverage of all points on the earth that are within view (that is, within the satellite footprint), would not waste power in directions where there are not users and would provide some level of gain so as to partly make up for power limitations in the transmitting equipment or sensitivity limitations in the receiving equipment. These considerations apply to uplink as well as downlink antenna pattern models.

II. Model of the ideal radiator for this situation

We begin with the assumption that all points on the earth are of equal importance in terms of being covered by RF from the satellite. Put another way, it is desired to equally illuminate all visible points on the ground below. It is assumed that the ground track and averaged "footprint" of the satellite are distributed about the earth so as to make the equal coverage property possible and desirable in terms of the user population.

The ideal antenna model presented here requires spacecraft stabilization in two axes such that one satellite facet can be kept pointing at earth at all times. The effect of this type of stabilization on spacecraft powering

has been discussed previously [1]. The effect on thermal performance is not considered in this study but should be addressed separately.

The radiation pattern considered is symmetric about the vertical axis which connects the center of the earth and the satellite. Rotation about the axis (changing the azimuth parameter) does not influence signal strength. For a fixed altitude, slant range to a point on the earth is a function of a single parameter, the angle between the segment joining the satellite with that point on the earth and the axis of symmetry.

Actual antenna and stabilization schemes will give patterns which are not symmetric about this axis. This does not necessarily mean that they are unacceptable as long as the variation is not so great that it causes signal levels to drop below desired margins or to fluctuate rapidly in time with azimuth changes due to satellite rotation or orbital motion. Such fluctuations could result in disruption of digital communications.

III. Ideal radiation pattern as a function of the angle to the nadir axis only

For a satellite in circular orbit, the parameters of interest are the altitude of the satellite and the radius of the earth.

R_e = radius of the earth, 6378.145 Km

R_s = distance, center of earth to satellite,

$$R_e + 800 = 7178.145$$

S = slant range, a dependent variable

a = angle from slant segment to vertical axis at satellite, a dependent variable

b = angle between user and satellite position at earth center

From trigonometry we have

$$S^2 = R_e^2 + R_s^2 - 2 R_e R_s \cos(b)$$

$$a = \arcsin[(R_e/S)\sin(b)]$$

and will always be < 90 degrees.

Special cases are S minimum where

$$S = R_s - R_e$$

$$a = b = 0 \text{ and}$$

S maximum where

$$S^2 = R_s^2 - R_e^2$$

$$b = \arccos(R_e/R_s)$$

$$a = 90 - b.$$

To provide equal illumination to all points in view (between S maximum and S minimum), apply the inverse square law

$$P = K/S^2$$

where

P is a relative power figure and
K is a constant of proportionality.

PTSE link margins specify 5 dB antenna gain for the worst case (longest distance) path so S max is referenced to this figure. The following table is produced from these equations.

Columns "A" and "accm %" are discussed below. A cross section of this pattern is plotted in Figure 1.

The ideal pattern radiates mostly toward the horizon which is, for this altitude, 62.7 degrees from the vertical axis. Gain inside 45 degrees from the vertical axis can be less than unity without compromising the system link margins. Any radiation more than 62.7 degrees from nadir is superfluous unless the system has users significantly above the surface of the earth (i.e., in orbit).

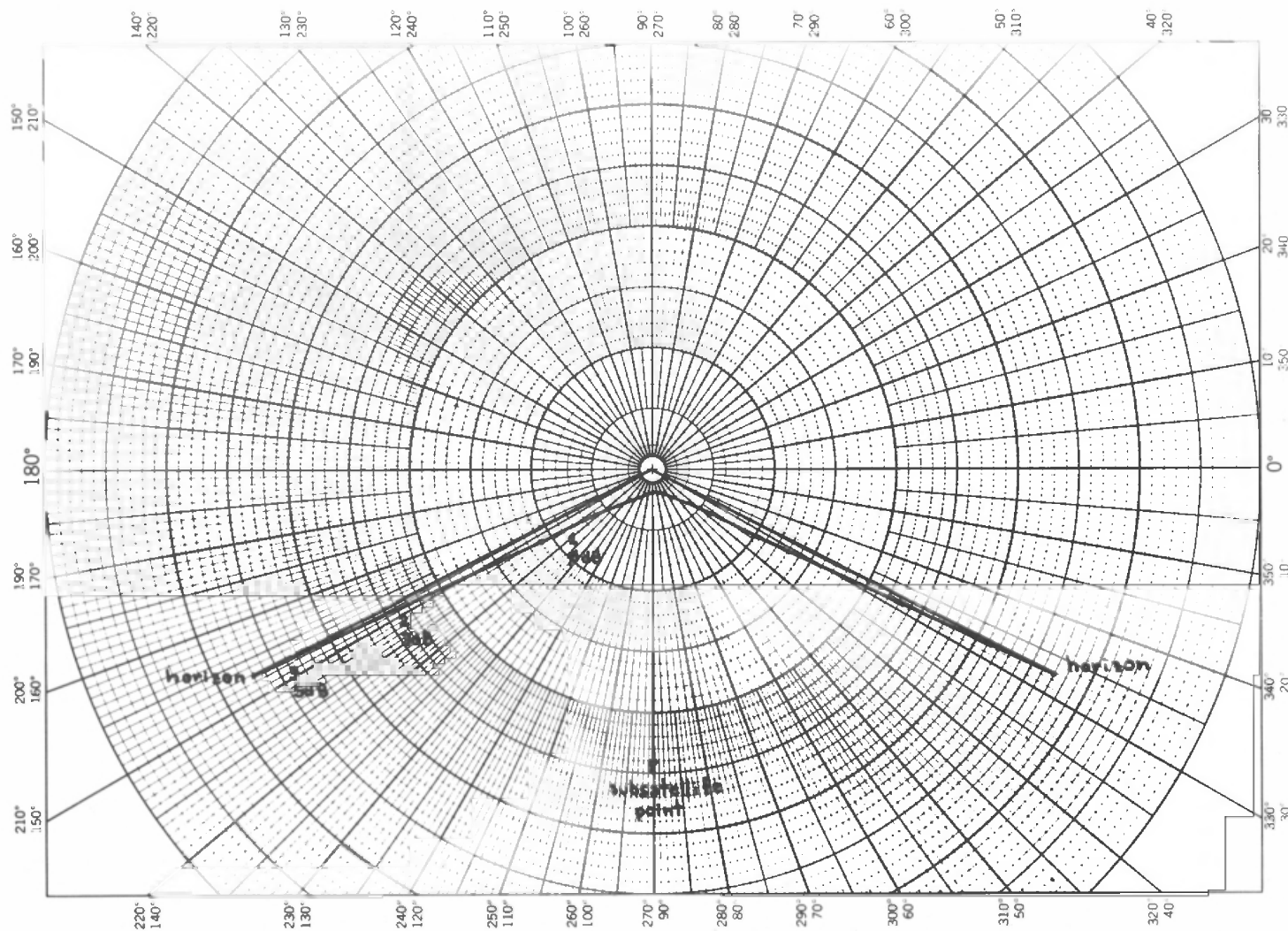


Figure 1: Cross-section of Satellite Pattern.

IV. Technique for comparing a real pattern to ideal pattern

It is physically impossible for an antenna to have exactly this idealized radiation pattern because of the discontinuity at the maximum radiation angle if for no other reason. Some antennas which may be realized in practice may perform adequately. A method of comparing an actual pattern to the idealized one is desired.

An explanation of the "A" and "accm" columns of Table 1 leads to an intuitive method of comparison. Refer to Figure 2.



Figure 2: Geometries.

The surface area of a spherical sector (or satellite footprint) is $(\pi \cdot D \cdot h)$ where

D is the diameter of the sphere (R_e) and

h is the depth of the slice at the center of the sector cut from the sphere. See Figure 2(a).

From trigonometry we have that

$$h = R_e \cdot (1 - \cos b)$$

using the variables introduced above.

For the entire footprint we therefore have

$$h = R_e \cdot (1 - R_e/R_s) \text{ and}$$

$$A_{\text{total}} = 2 \cdot \pi \cdot R_e^2 \cdot (1 - R_e/R_s)$$

The "A" values in the table are the surface areas of rings in the footprint defined by the previous radius "a" to the current radius "a" expressed as a fraction of the total footprint area. See Figure 2(b). The total footprint area in square kilometers is given at the bottom.

The "accm" value is simply the accumulated area of the surface circle described by the current "a." See Figure 2(c).

These numbers are used only to estimate the relative importance of antenna performance in the direction in question. The argument is that it is more important to

Table 1. Radiation pattern for the ideal PTSE antenna

a	S	P	dB	b	A %	accm %
0.0	800.0	0.2	-7.3	0.0	0.0	0.0
7.9	808.7	0.2	-7.2	1.0	0.1	0.1
15.5	834.1	0.2	-6.9	2.0	0.4	0.5
22.4	874.9	0.2	-6.5	3.0	0.7	1.2
28.6	929.0	0.3	-6.0	4.0	1.0	2.2
34.0	994.2	0.3	-5.4	5.0	1.2	3.4
38.6	1068.5	0.3	-4.8	6.0	1.5	4.9
42.5	1150.0	0.4	-4.1	7.0	1.8	6.7
45.8	1237.4	0.4	-3.5	8.0	2.0	8.7
48.6	1329.4	0.5	-2.9	9.0	2.3	11.0
51.0	1425.2	0.6	-2.3	10.0	2.6	13.6
53.0	1523.9	0.7	-1.7	11.0	2.9	16.5
54.7	1625.1	0.8	-1.1	12.0	3.1	19.6
56.1	1728.2	0.9	-0.6	13.0	3.4	23.0
57.3	1833.0	1.0	-0.1	14.0	3.7	26.7
58.4	1939.1	1.1	0.4	15.0	3.9	30.6
59.2	2046.2	1.2	0.9	16.0	4.2	34.8
60.0	2154.3	1.4	1.3	17.0	4.4	39.2
60.6	2263.1	1.5	1.7	18.0	4.7	43.9
61.1	2372.5	1.6	2.2	19.0	5.0	48.9
61.5	2482.4	1.8	2.5	20.0	5.2	54.1
61.8	2592.6	2.0	2.9	21.0	5.5	59.6
62.1	2703.2	2.1	3.3	22.0	5.7	65.3
62.3	2814.1	2.3	3.6	23.0	6.0	71.3
62.5	2925.1	2.5	4.0	24.0	6.2	77.6
62.6	3036.3	2.7	4.3	25.0	6.5	84.1
62.7	3147.6	2.9	4.6	26.0	6.7	90.8
62.7	3258.9	3.1	4.9	27.0	7.0	97.8
62.7	3293.2	3.2	5.0	27.3	2.2	100.0

28,486,980 Km² total area of footprint

meet desired antenna specifications in a subsector containing more surface area and therefore, on the average, more users. The "A" and "accm" columns are therefore called the "importance columns."

From the importance columns, it is seen that the right cone (all angles inside 45 degrees from the local vertical axis) comprises only ten percent of the surface area of the footprint! It is also seen that 50 percent of the area of the footprint occurs in the last 1.5 degrees of the idealized antenna pattern. Clearly, the circular lobe toward the horizon should be greatly emphasized in antenna design priority. The area in view directly below the satellite is relatively unimportant and requires relatively little gain in any case.

V. PTSE user antenna pattern

In order to produce the same table but in terms of user elevations at ground level, this formula is introduced $el = 90 - \arcsin[(R_s/S) \cdot \sin(b)]$ in which the desired result is always < 90 .

Here, "A" and "accm" values refer to the amount of time that the satellite spends at the elevation in question as a percentage of total time above the horizon.

The satellite is seen spending less than ten percent of its time in view above 35 degrees elevation, and more than 55 percent below ten degrees. Also, most of the user antenna's gain should be concentrated within ten degrees of the horizon because greatest range to the satellite occurs at these elevations. Although the antenna needs to have some radiation above 35 degrees, it is not very important in terms of satellite access time and the gain does not need to be substantial or even near unity for adequate operation.

VI. Practical considerations

The "ideal" antenna pattern would not be so ideal in practice since the proposed satellite cannot be perfectly stabilized to an apparently stationary condition with respect to the earth. A torus shaped lobe extending toward the horizon must be at least ten or twenty degrees wide (in "a") so as to allow for attitude errors.

A separate command receiver antenna, and possibly a telemetry transmitting antenna should have some radiation in all directions so that before attitude stabilization, or in case of loss of stabilization, the spacecraft can receive and transmit to at least a well equipped com-

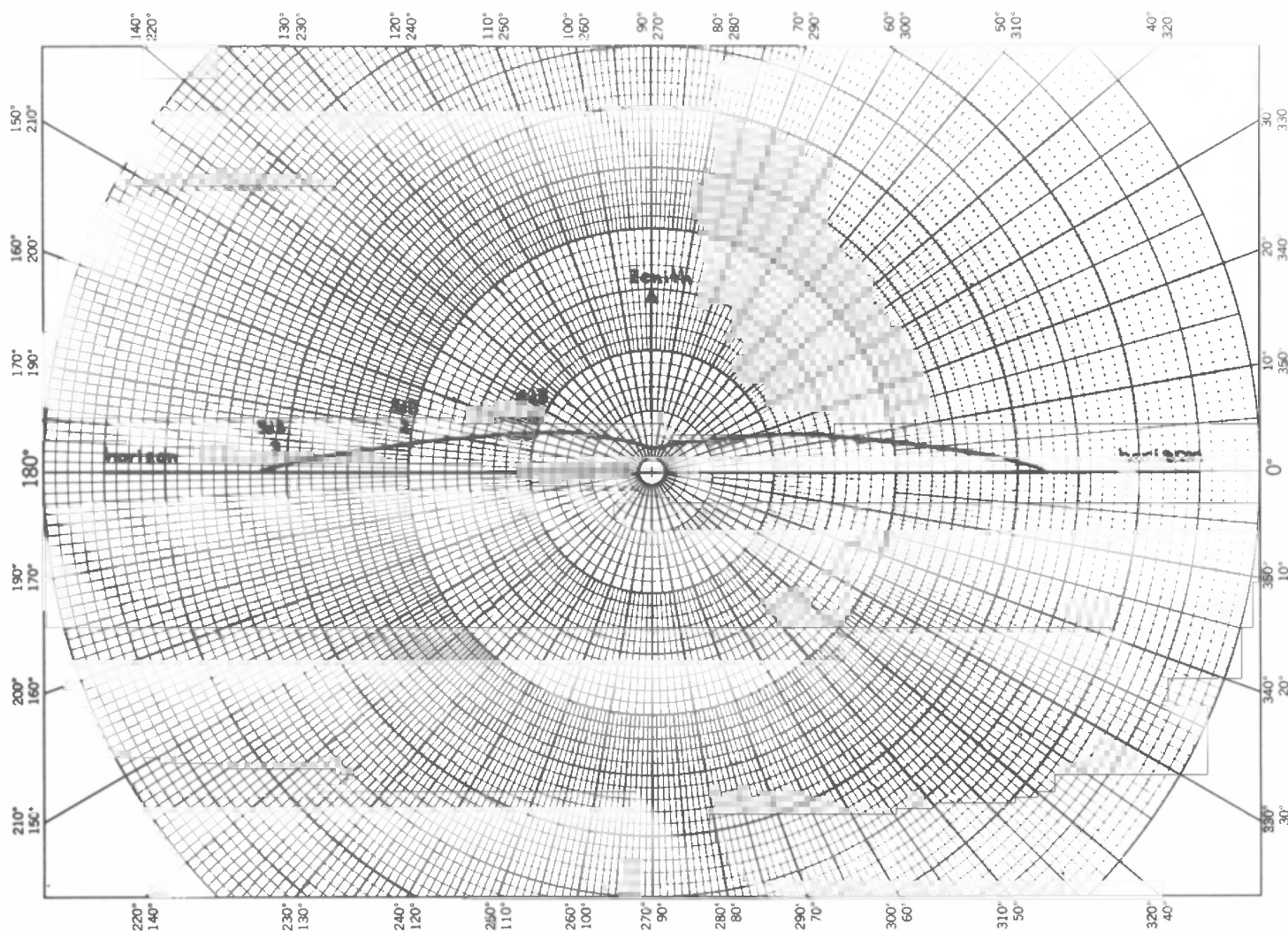


Figure 3: Cross-section of User Antenna Pattern.

mand station. The antenna pattern just developed considers primarily the nominal user station under nominal operational conditions.

An attempt to achieve this radiation pattern could rule out the possibility of circular polarization. Since the spacecraft is to be stabilized in two axes, however, this should not be a problem, in fact, it works in favor of the fixed antenna user. Polarization as seen from near the horizon would be either vertical or horizontal. If a "vertical" antenna, one parallel to the vertical axis at the satellite, is used, polarization would be seen as vertical from any ground vantage. If horizontal because of some sort of dipole or turnstile arrangement at the satellite, polarization as seen from the ground would be horizontal from the horizon, circular directly below, and elliptical (with the linear component being horizontal) at intermediate points.

VII. Qualitative discussion of antenna candidates

In satellites that are not stabilized with respect to the geocenter, an isopole is considered ideal. By definition, the gain in all directions is equal and unity. This is adequate in terms of the model just developed except for the five degree "thick" critical area around 60 degrees from the vertical axis that contains 75% of the important coverage. In this region, the isopole is between 30%

and 100% adequate. An estimate of the its merit is about 40%.

A "vertical" type whip which by means of mounting or radial geometry could be made to concentrate most of its energy at the critical angle at all azimuths could be very satisfactory. In attempting to "bend" maximum radiation angle down from the local horizontal plane, modifications to the size and shape of the ground plane and distance of the driven element from it should be considered theoretically and experimentally.

A beam like helix or crossed yagi antenna with 5 dB forward gain will have a half power beamwidth of around 100 degrees (50 degrees each side of center). Such a beam placed symmetrically within the desired radiation pattern, that is, with the centerline on the vertical axis, will have most of its gain where it is not needed and where it is not very important statistically. The gain will be falling away drastically just at the angles where it does become very important. Such a beam, or even a more directive one with higher gain, would do well if mounted at the critical angle aiming toward the horizon but its radiation properties would then have azimuth dependencies. Perhaps four or more of them around the "bottom" of the satellite could have satisfactory, overlapping coverage if the feeds could be phased properly.

Table 2. Radiation pattern for the ideal PTSE user antenna

el	S	P	dB	b	A %	accm %
90.0	800.0	0.2	-7.3	0.0	0.0	0.0
81.1	808.7	0.2	-7.2	1.0	0.1	0.1
72.5	834.1	0.2	-6.9	2.0	0.4	0.5
64.6	874.9	0.2	-6.5	3.0	0.7	1.2
57.4	929.0	0.3	-6.0	4.0	1.0	2.2
51.0	994.2	0.3	-5.4	5.0	1.2	3.4
45.4	1068.5	0.3	-4.8	6.0	1.5	4.9
40.5	1150.0	0.4	-4.1	7.0	1.8	6.7
36.2	1237.4	0.4	-3.5	8.0	2.0	8.7
32.4	1329.4	0.5	-2.9	9.0	2.3	11.0
29.0	1425.2	0.6	-2.3	10.0	2.6	13.6
26.0	1523.9	0.7	-1.7	11.0	2.9	16.5
23.3	1625.1	0.8	-1.1	12.0	3.1	19.6
20.9	1728.2	0.9	-0.6	13.0	3.4	23.0
18.7	1833.0	1.0	-0.1	14.0	3.7	26.7
16.6	1939.1	1.1	0.4	15.0	3.9	30.6
14.8	2046.2	1.2	0.9	16.0	4.2	34.8
13.0	2154.3	1.4	1.3	17.0	4.4	39.2
11.4	2263.1	1.5	1.7	18.0	4.7	43.9
9.9	2372.5	1.6	2.2	19.0	5.0	48.9
8.5	2482.4	1.8	2.5	20.0	5.2	54.1
7.2	2592.6	2.0	2.9	21.0	5.5	59.6
5.9	2703.2	2.1	3.3	22.0	5.7	65.3
4.7	2814.1	2.3	3.6	23.0	6.0	71.3
3.5	2925.1	2.5	4.0	24.0	6.2	77.6
2.4	3036.3	2.7	4.3	25.0	6.5	84.1
1.3	3147.6	2.9	4.6	26.0	6.7	90.8
0.3	3258.9	3.1	4.9	27.0	7.0	97.8
0.0	3293.2	3.2	5.0	27.3	7.2	100.0

A turnstile placed $3/8$ wave above a ground plane has a pattern which is shown in Figure 6-27 B of the *Satellite Experimenters Handbook* (3rd printing, 1985) which matches PTSE user requirements closely if the main lobes can contain 5 dB gain. Notice that the ground plane itself should extend beyond the driven elements at least a half wave in each direction. This would be a problem at the satellite for a two meter antenna and could only be achieved marginally at 70 cm without radial extensions. Turnstiles are traditionally made from crossed dipoles. Some variation in the radiation pattern with azimuth will be present although it is likely to be acceptably small.

VIII. Conclusions

The requirement of a low earth orbiting satellite antenna to adequately cover all of its footprint at any time is clear. Whether the requirement corresponds to an antenna and satellite control situation that can actually be achieved with amateur radio resources is unknown. An adequate user antenna will be simple and inexpensive. Equipment that amateurs already have, not previously intended for satellite work, will in many cases be ample to the task of nominal network participation.

This paper does not attempt to report on construction or testing of an actual satellite antenna or spaceframe integration and testing. The intention here has been to consider satellite user requirements and stimulate physically meaningful goals for the forthcoming system design effort.

References

1. Duncan, C., "Power Budget and Eclipse Considerations for PTSE-H." *AMSAT-NA Technical Journal*, Volume 1, No. 1.

IX. Comments on the idealized pattern for other orbital altitudes

Tables for satellites at different altitudes were generated and studied. A few comments and comparisons may be of interest.

300 Km satellite altitude

The requirement for any gain overhead nearly vanishes; it is 16 dB below the power gain needed at the horizon. The area of importance is much sharper, 50% of the footprint area is concentrated between 71.8 and 72.8 degrees from vertical. The footprint contains less than half of the area of the 800 Km altitude footprint.

600 Km satellite altitude

This case is not practically different from the 800 Km pattern.

1500 Km satellite altitude

This case is also similar to 800 Km except that 50% of coverage is within a two degree slice of the antenna pattern. The total surface area in the footprint is 70% greater.

35,700 Km geosynchronous altitude

From this altitude, nearly half of the surface of the earth is covered. A 20 dB gain antenna boresite on the apparent center of the earth would provide adequate gain to all points visible. This case is easier to visualize considering the way that we normally perceive the earth as viewed from space.

The Integrated Housekeeping Unit—A Method of Telemetry, Command and Control for Small Spacecraft

by Gordon Hardman, KE3D

(Presented at the 1st Utah State University Conference)
(on Small Satellites, October 7, 1987)

ABSTRACT

In a small satellite environment where the total power budget for telemetry, command, ranging and all satellite control functions is not to exceed 2 Watts, rather focused measures must be taken to consolidate these functions. Since 1975 AMSAT has been developing the concept of the Integrated Housekeeping Unit (IHU) for this purpose. The IHU combines the traditional telemetry encoder and command decoder with a multi-tasking microcomputer so that this single module is capable of handling all spacecraft functions simultaneously. The nature of the orbit planned for the series of spacecraft for which this unit was developed results in a high radiation environment for the IHU. Special attention was paid to this in the design. A high-level language has been developed for the IHU. Called IPS, it operates on up to eight simultaneous tasks, including telemetry and command processing, and navigational functions concerned with orbital insertion and stationkeeping. The development of this concept is described in this paper.

1.0 MISSION OUTLINE

In 1975 AMSAT began planning the first in a series of high altitude communications satellites for the amateur radio community. Called Phase 3, the primary payload function was to provide one or more medium bandwidth linear transponders for general communications by licensed amateur radio operators. Frequency division multiple access (FDMA) allowed the available bandwidth to be shared among many simultaneous users.

An elliptical Moniya-type orbit was chosen (See Figure 1) with a perigee of about 1500 km and an apogee of 36000 km. With a period of about 700 minutes, and an inclination 63 degrees this orbit has the property that the argument of perigee remains almost fixed. The two daily apogees thus occur above the Northern hemisphere, in view of those parts of the earth with the greatest concentrations of potential users. A three year mission lifetime was the aim.

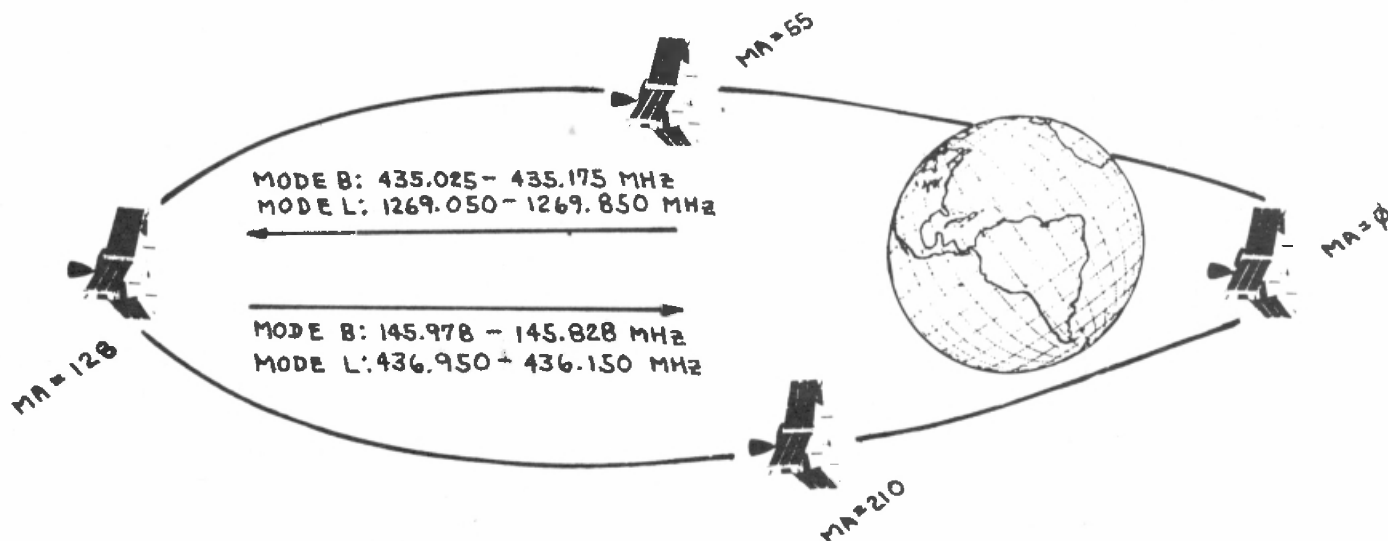


Figure 1: Elliptical Orbit of Phase III-B.

2.0 SPACECRAFT DESCRIPTION

The spacecraft is a three-armed structure. (See Figure 2). It is spin stabilized, spin rate and direction being controlled by a system of magnetic torquing coils which are activated near perigee, when the Earth's magnetic field is strongest.

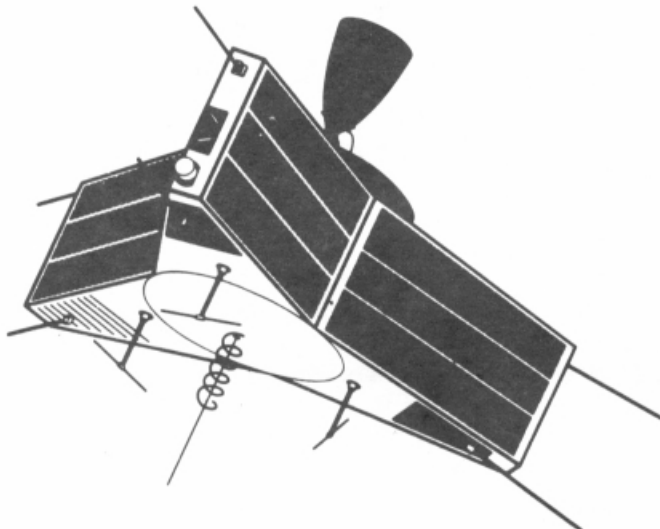


Figure 2: Phase III-B Spacecraft

Launch into an initial geostationary transfer orbit was selected as being compatible with a wide range of other missions, thus making effective use of the dual launch capabilities of vehicles such as Ariane. In order to get into a Molniya orbit, an apogee kick motor was included to raise the perigee and increase the inclination of the orbit.

The spacecraft has two sets of antennas; high gain for use near apogee, and low gain for use near perigee, since the high gain antennas are then pointing away from the Earth.

A functional block diagram of the Phase-3B spacecraft is shown in Figure 3. This was the second of the three spacecraft so far constructed, and is a good example of the series. There are five major systems; each will be described briefly.

2.1 Power System

Six solar panels provide a beginning-of-life power of about 50 Watts. A novel redundant battery-charge-regulator (BCR) keeps an 8AH battery charged. An auxiliary battery is maintained offline in a discharged state. The 14 Volt battery voltage is available for use directly by some modules; and a 10 Volt regulated supply is also distributed. A separation switch allows power to the transmitters only once separation from the launch vehicle has occurred, to prevent radiation of RF energy and possible interference with launch vehicle systems during launch.

2.2 Communication Systems

Two linear transponders are provided. Only one can be active at any given time due to DC power limitations; additionally, on Phase-3B, the input band of one transponder is close to the output band of the other, making them incompatible from an interference point of view. Each transponder can feed either a high or low gain antenna system, as mentioned above.

2.3 Attitude Control System

The attitude control system is based around the sensor/electronics unit (SEU). Pulses from Sun and Earth sensors are pre-processed in the SEU and then passed on to the IHU, which uses them to update its navigational information. Near perigee, the IHU commands power control circuits in the SEU to pulse the torquing coils as needed to maintain desired spin rate and direction.

2.4 Motor System

Phase-3B employed a 400 Newton bipropellant thruster [1]. Efficient use of the re-start capabilities of this motor was planned, allowing the desired orbit to be obtained by at least two burns. The liquid ignition unit (LIU) controlled the motor; it in turn was commanded by the IHU.

3.0 IHU FUNCTIONS

At the heart of the spacecraft is the IHU (Fig. 3). Once the spacecraft is on-orbit, it performs the following functions:

- Collection of data from 64 analog telemetry channels
- Generation of two separate telemetry streams
- Processing of all uplinked command information after command validation
- Processing of all sensor data from the SEU
- Execution of magnet torquing maneuvers to change spacecraft attitude (operates closed loop with sensors)
- Monitoring of the power status of the spacecraft and manipulation of the BCR when required. (under closed loop control)
- Execution of a schedule program which modifies the operating modes of the two transponders and four antennas at different points along the orbit track

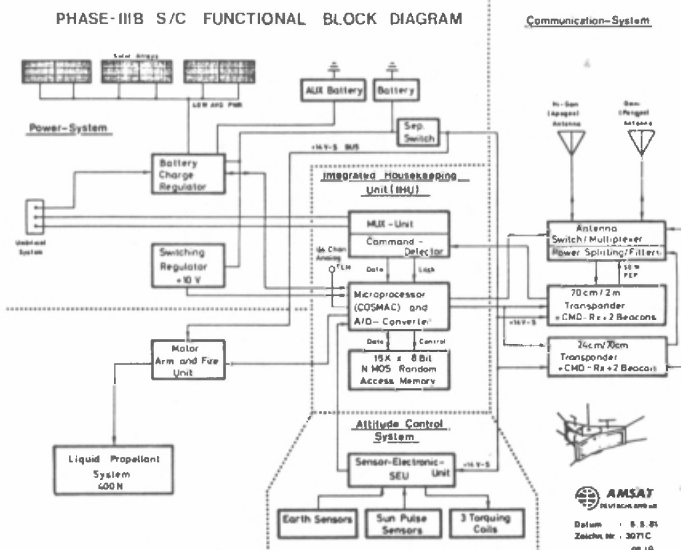


Figure 3: Functional Block Diagram

- Management of a bulletin board within the spacecraft to allow command stations to advise one another on the health of the spacecraft and the last commands sent.
- Performance of other miscellaneous tasks

During the orbit-insertion phase, the IHU performs most of the above tasks, except that the transponders are not available for general use, and are only to be enabled to allow ranging to take place. During this phase the IHU must:

- Prepare the LIU for operation, by operating safety interlocks which pressurize the motor systems from a master helium tank
- Orient the spacecraft spin vector correctly for each burn
- Command the LIU to fire the motor at the correct time and for the correct duration

4.0 IHU HARDWARE

The IHU is comprised of four circuit boards, arranged in three layers, as follows:

Inner layer: Random access memory (RAM) board
 Middle layer: Central processing unit (CPU) board
 Outer layer: Multiplexer (MUX) and command detector (CMD) boards

4.1 CPU board

The RCA COSMAC CDP1802 microprocessor was selected as the processor, primarily because it was the only suitable CMOS device available at the time of the initial design. This choice has proven to be a good one, as the processor has been flexible and powerful enough to meet the more complex demands of the later missions. In addition, a radiation hardened version of the processor, built using silicon-on-sapphire technology became available.

The CPU runs at a clock rate of 1.68 MHz, and the card contains all the support circuitry for the 1802. Additionally, there are 5 general purpose latched 8-bit output ports; 4 general purpose 8-bit input ports; an 8-bit digital-to-analog converter; and circuitry to generate two separate serial data streams for telemetry beacons.

A novel feature of this design is that the IHU uses only RAM. There is no read-only-memory (ROM). Re-booting of the computer is accomplished by circuitry which senses a particular sequence of bits in the command uplink data stream, and then places the processor in the "Load" mode. In this mode, the 1802 simply resets a memory address pointer to zero, and performs a memory save operation on incoming data bytes, incrementing the address pointer after each operation. Thus, the bytes in the first valid command block after a reset are stored in sequential locations in low memory. After 128 bytes (1 command block) the processor is put back into "Run" mode, and begins execution with memory address 0. Thus, the first 128 byte block must contain a minimal bootstrap loader which will then load subsequent blocks in the appropriate memory locations until the entire operating system has been loaded, and control can be passed to it.

The reason for this somewhat unusual approach is that at the time of the initial design there were doubts that ROM then available would be able to survive the radiation expected in the proposed orbit. In fact, no problems have been experienced with this approach, which also possesses the property that the bootstrap loader could be changed in orbit as operational experience dictates.

4.2 RAM board

A computer memory of 2k bytes was envisaged early in the design cycle, but by the time the first flight unit had been built, this had grown to 16k. The latest unit completed had 32k of radiation-hardened memory.

The problem of radiation damage is explicitly attacked at the electrical design stage of the memory. An error-detection-and-correction (EDAC) scheme is used. The processor operates with 8-bit bytes; as a memory-save operation takes place, the byte to be saved passes through an encoding matrix which generates 4 additional bits. These are stored in memory at the same address as the main byte. On being retrieved from memory the twelve bits pass through another matrix which can detect a single-bit error, and tell in which bit it occurred, thus allowing it to be corrected. The processor, as a background task, is continuously cycling through the memory address space, reading each byte, and then writing it back. If a radiation induced bit-flip has occurred, it will be corrected during the read cycle, and the correct byte will be restored during the write cycle. The cycle time for this process is about 43 seconds for a 16k memory. The memory is thus protected against radiation-induced bit-flips, provided that no more than one occurs in a word in about a 1 minute period.

4.3 MUX and CMD

The outermost layer of the IHU contains these two boards. The CMD is fed a differentially coded serial NRZ data stream, from the command receiver and performs bit timing recovery and differential data extraction. The CMD monitors the uplink, and on receipt of a valid unique flag, begins passing data to the CPU in a byte-parallel format. The 128 bytes following the flag represent an uplink "packet."

The MUX is a 64 way analog switch. This converts the single DAC on the CPU into a 64 channel scanning voltmeter. The input range is 0 to $\sqrt{2}$ volts, so each quantity to be measured must be converted to lie in this range. For voltages, this is easily achieved. Currents are measured by using a "magnetometer" approach: sensing the bias in the hysteresis loop of a metallic core through which a wire carrying the current to be measured is passing. Temperatures are measured by thermistors, which are biased to produce a voltage in the correct range over temperature.

5.0 Radiation Protection

A spacecraft in a Molniya type of orbit encounters the Van Allen radiation belts twice per orbit. Measures must be taken to ensure survival of the spacecraft electronics in the face of this large cumulative radiation dose.

It was determined early in the design that the most susceptible module to radiation would be the memory. This was the subject of special attention, which included the following measures, in addition to the EDAC design described above:

- Choosing RAM chips with the best possible radiation properties.
- Self-shielding, by placing the RAM board as close to the center of an arm as possible, thus using outer modules as shields.
- Local shielding. Each integrated circuit was sandwiched between two small rectangular sheets of tantulum.
- Group shielding. The RAM ICs were grouped together on the circuit board, and the whole group surrounded by a brass enclosure.

6.0 Software

Special purpose software was developed for the IHU. The main concern was to ensure that the spacecraft was capable of operation for extended periods out of view of a command station. This situation occurs because of the nature of the orbit, and the relatively small number of command stations and their geographic distribution.

The IHU runs a high-level language called IPS (Interpreter for Process Structures), a threaded-coded language similar to Forth. In this language, a number of basic procedures are defined, and applications are built up by stringing together addresses of these routines in the order in which they are to be executed. Each application is given a new name, which can be used in subsequent definitions. The language is thus powerful, flexible, and extensible. An additional important consideration is that it is extremely efficient in terms of memory usage, making good use of a scarce spacecraft resource.

Execution of multiple tasks is made possible by placing them in a special "chain." This is a circular list of very high level definitions to be executed. The IHU sequences through them executing each in term. A task can "dechain" itself, leaving a no-op in its place; or it can "enchain" another task in its place. There are eight chain positions available for application tasks.

The IHU maintains a real time clock, and also contains an ephemeris which allows autonomous operation of the attitude control system. This is possible because the IHU also has a simple model of the geomagnetic field; this combined with orbital information allows the torquing coils to be energized at the proper times.

7.0 Performance Results

The IHU development effort is briefly outlined in Table 1. The major change over time has been in the memory. Early units used NMOS RAM chips; the last unit built used CMOS. This change has had several important benefits:

- Power consumption is less
- Memory size has been doubled to 32k
- Radiation hardness has been increased

The CMOS unit is currently undergoing qualification, and is expected to become the prime IHU for the Phase 3-C mission when this is completed.

The only unit to see service in orbit so far has been that on the Phase 3-B spacecraft (Oscar 10). This unit delivered nearly four years of satisfactory service before radiation damage reduced its usefulness. The unit is now only conditionally operational, as it is difficult to maintain large programs operational in memory.

8.0 Lessons Learned

The IHU concept has been remarkably successful. The design is still relatively stable 12 years after its conception. It has proved powerful and flexible enough to grow with the mission complexity.

It has also been responsible for bringing a certain amount of order to the interface control process. Since the prime interface of most modules is with the IHU, a document called "spacecraft as seen by the computer" was drawn up early in each mission. This detailed what was connected to each port, and how it was expected to respond in any situation. Thus, development of other spacecraft systems could proceed in parallel with the software development.

One problem that has been recognized is that as the spacecraft complexity grows, the sheer number of wires being brought into the IHU grows. This is because each control signal is fully decoded in the IHU. The last of the present series of IHUs has been built, and for future missions [2] Amsat is studying the possibility of a bus-oriented control hierarchy. This will allow the use of a common IHU in different missions of widely varying complexity, with only minor variations in the wiring harness.

The author would like to recognize the work of the following people who participated in the IHU development project: Steve Robinson, W2FPY; Karl Meinzer, DJ4ZC; Jan King, W3GEY; Andy Deskur, KAIM; Vern Riportella, WA2LQQ; and Ron Dunbar, W0PN.

Model	Memory	Deployment
Conceptual design	2k NMOS	N/A
Phase 3-A	16k NMOS	Lost (Launcher failure)
Phase 3-B	16k NMOS	1983-1987
Phase 3-C prime	16k NMOS	Awaiting launch in early 1988
Phase 3-C altern.	32k CMOS	In qual. May become prime

Table 1. IHU genealogy.

References

[1] Richard L. Daniels, "'Bare Bones' Propulsion for Small, Low-cost Satellites," 1st USU Conf. on Small Satellites, Logan UT, 1987.

[2] Dick Jansson, "Spacecraft Technology Trends in the Amateur Satellite Service," 1st USU Conf. on Small Satellites, Logan UT, 1987.

Keplerian Elements for Weather Satellites

Satellite: meteor 2-14
Catalog number: 16735

Epoch time: 88022.16781315
Element set: 212
Inclination: 82.5409 deg
RA of node: 293.4003 deg
Eccentricity: 0.0015230
Arg of perigee: 12.5447 deg
Mean anomaly: 347.6109 deg
Mean motion: 13.83770263 rev/day
Decay rate: 6.0e-08 rev/day²
Epoch rev: 8364

Satellite: meteor 2-15
Catalog number: 17290

Epoch time: 88022.02546301
Element set: 136
Inclination: 82.4663 deg
RA of node: 204.5031 deg
Eccentricity: 0.0012018
Arg of perigee: 256.0233 deg
Mean anomaly: 103.9598 deg
Mean motion: 13.83577001 rev/day
Decay rate: 6.0e-08 rev/day²
Epoch rev: 5279

Satellite: meteor 2-16
Catalog number: 18312

Epoch time: 88022.04614306
Element set: 81
Inclination: 82.5567 deg
RA of node: 265.2512 deg
Eccentricity: 0.0011653
Arg of perigee: 187.5321 deg
Mean anomaly: 172.5666 deg
Mean motion: 13.83334672 rev/day
Decay rate: 5.3e-07 rev/day²
Epoch rev: 2170

Satellite: meteor 3-1
Catalog number: 16191

Epoch time: 88022.50786552
Element set: 729
Inclination: 82.5500 deg
RA of node: 237.5442 deg
Eccentricity: 0.0019701
Arg of perigee: 132.8655 deg
Mean anomaly: 227.4052 deg
Mean motion: 13.16925085 rev/day
Decay rate: 4.3e-07 rev/day²
Epoch rev: 10817

Satellite: noaa-9
Catalog number: 15427

Epoch time: 88030.92320376
Element set: 233
Inclination: 99.0830 deg
RA of node: 3.8634 deg
Eccentricity: 0.0014729
Arg of perigee: 287.9653 deg
Mean anomaly: 71.9905 deg
Mean motion: 14.11551356 rev/day
Decay rate: 3.5e-07 rev/day²
Epoch rev: 16144

Satellite: noaa-10
Catalog number: 16969

Epoch time: 88027.71773820
Element set: 112
Inclination: 98.6948 deg
RA of node: 60.5252 deg
Eccentricity: 0.0012909
Arg of perigee: 280.0554 deg
Mean anomaly: 79.9162 deg
Mean motion: 14.22543120 rev/day
Decay rate: 2.04e-06 rev/day²
Epoch rev: 7066